NASA CR-132575-4

SUBSTANTIATING DATA FOR

ARROW-WING SUPERSONIC CRUISE AIRCRAFT STRUCTURAL DESIGN CONCEPTS EVALUATION

Volume 4, Sections 15 through 21

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by

I. F. Sakata and G. W. Davis

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Lockheed-California Company
Burbank, California

for Langley Research Center

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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SUMMARY

An analytical study was performed to determine the structural approach best suited for the design of a Mach 2.7 arrow-wing supersonic cruise aircraft.

Results, procedures, and principal justification of results are presented in Reference 1. Detailed substantiation data are given herein. In general, each major analysis is presented sequentially in separate sections to provide continuity in the flow of the design concepts analysis effort. In addition to the design concepts evaluation and the detailed engineering design analyses, supporting tasks encompassing: (1) the controls system development (2) the propulsion-airframe integration study, and (3) the advanced technology assessment are presented.

Reference 1 Sakata, I. F. and Davis, G. W.: Evaluation of Structural Design Concepts for an Arrow-Wing Supersonic Cruise Aircraft NASA CR- 1976

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INTRODUCTION

The design of an economically viable supersonic cruise aircraft requires reduced structural mass fractions attainable through application of new materials, advanced concepts and design tools. Configurations, such as the arrow-wing, show promise from the aerodynamic standpoint; however, detailed structural design studies are needed to determine the feasibility of constructing this type of aircraft with sufficiently low structural mass fraction.

For the past several years, the NASA Langley Research Center has been pursuing a supersonic cruise aircraft research program (1) to provide an expanded technology base for future supersonic aircraft, (2) to provide the data needed to assess the environmental and economic impacts on the United States of present and especially future foreign supersonic cruise aircraft, and (3) to provide a sound technical basis for any future consideration that may be given by the United States to the development of an environmentally acceptable and economically viable commercial supersonic cruise aircraft.

The analytical study, reported herein, was performed to provide data to support the selection of the best structural concept for the design of a supersonic cruise aircraft wing and fuselage primary structure considering near-term start-of-design technology. A spectrum of structural approaches for primary structure design that has found application or had been proposed for supersonic aircraft design; such as the Anglo-French Concorde supersonic transport, the Mach 3.0-plus Lockheed F-12 and the proposed Lockheed L-2000 and Boeing B-2707 supersonic transports were systematically evaluated for the given configuration and environmental criteria.

The study objectives were achieved through a systematic program involving the interactions between the various disciplines as shown in Figures A through C. These figures present an overview of the study effort and provides a summary statement of work, as follows:

(1) Task I - Analytical Design Studies (Figure A).- This initial task involved a study wherein a large number of candidate structure

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concepts were investigated and subjected to a systematic evaluation process to determine the most promising concepts. An airplane configuration refinement investigation, including propulsion-airframe integration study were concurrently performed.

(2) Task II - Engineering Design/Analyses (Figure B).- The most promising concepts were analyzed assuming near-term start-of-design technology, critical design conditions and requirements identified, and construction details and mass estimates determined for the Final Design airplane. Concurrently, the impact of advanced technology on supersonic cruise aircraft design was explored.

(3) Task III - Mass Sensitivity Studies (Figure C).- Starting with the Final Design airplane numerous sensitivity studies were performed. The results of these investigations and the design studies (Task I and Task II) identified opportunities for structural mass reduction and needed research and technology to achieve the objectives of reduced structural mass.

Displayed on the figures are the time-sequence and flow of data between disciplines and the reason for the make-up of the series of sections presented in this report. The various sections are independent of each other, except as specifically noted. Results of this structural evaluation are reported in Reference 1. This reference also includes the procedures and principal justification of results, whereas this report gives detailed substantiation of the results in Reference 1. This report is bound as four separate volumes.

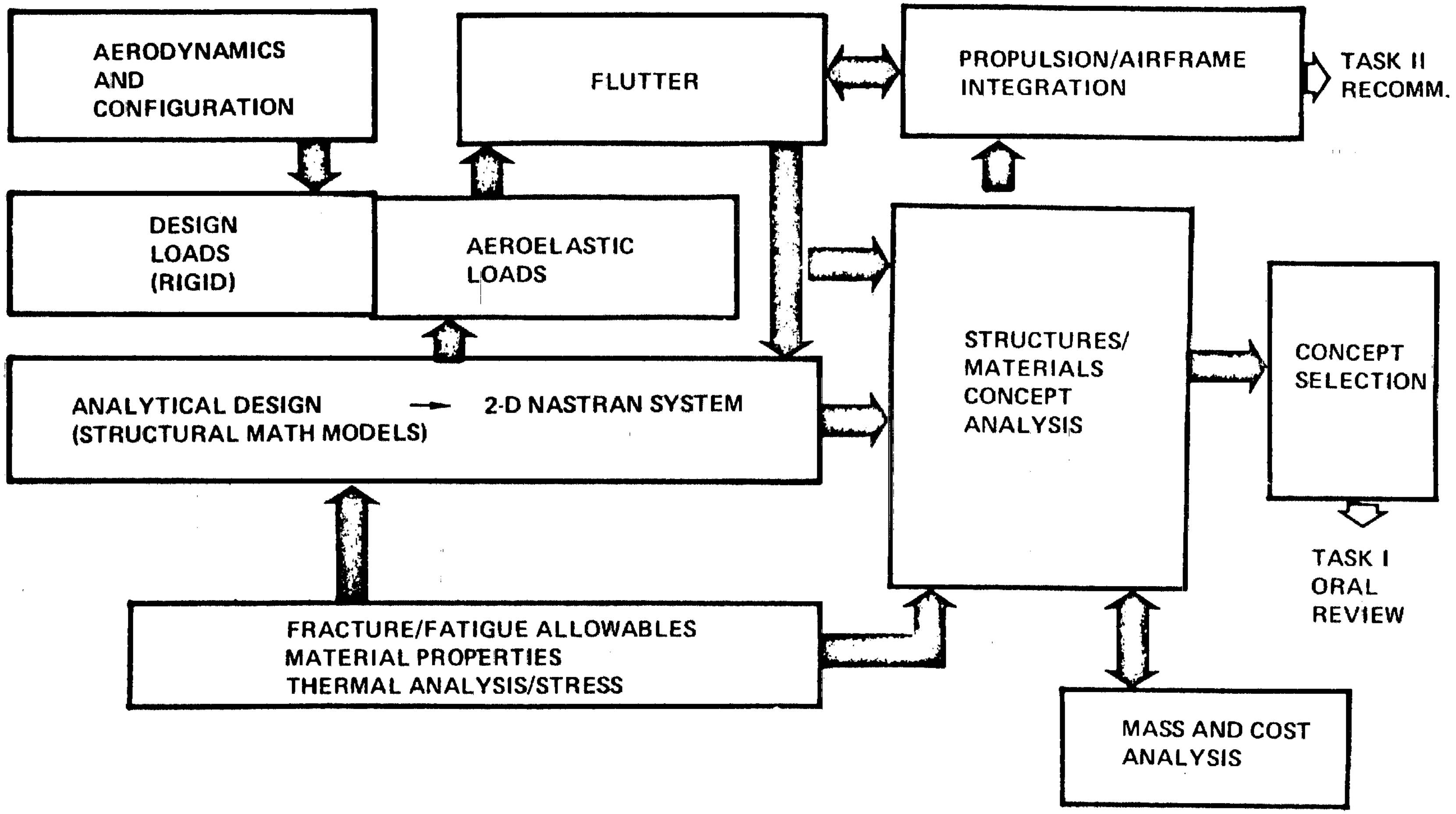
AERODYNAMICS AND CONFIGURATION
FLUTTER
PROPULSION/AIRFRAME INTEGRATION
TASK II RECOMM.
DESIGN LOADS (RIGID)
AEROELASTIC LOADS
STRUCTURES/ MATERIALS CONCEPT ANALYSIS
CONCEPT SELECTION
ANALYTICAL DESIGN (STRUCTURAL MATH MODELS)
→
2-D NASTRAN SYSTEM
TASK I ORAL REVIEW
FRACTURE/FATIGUE ALLOWABLES
MATERIAL PROPERTIES
THERMAL ANALYSIS/STRESS
MASS AND COST ANALYSIS

Figure A. Analytical Design Studies - Task I

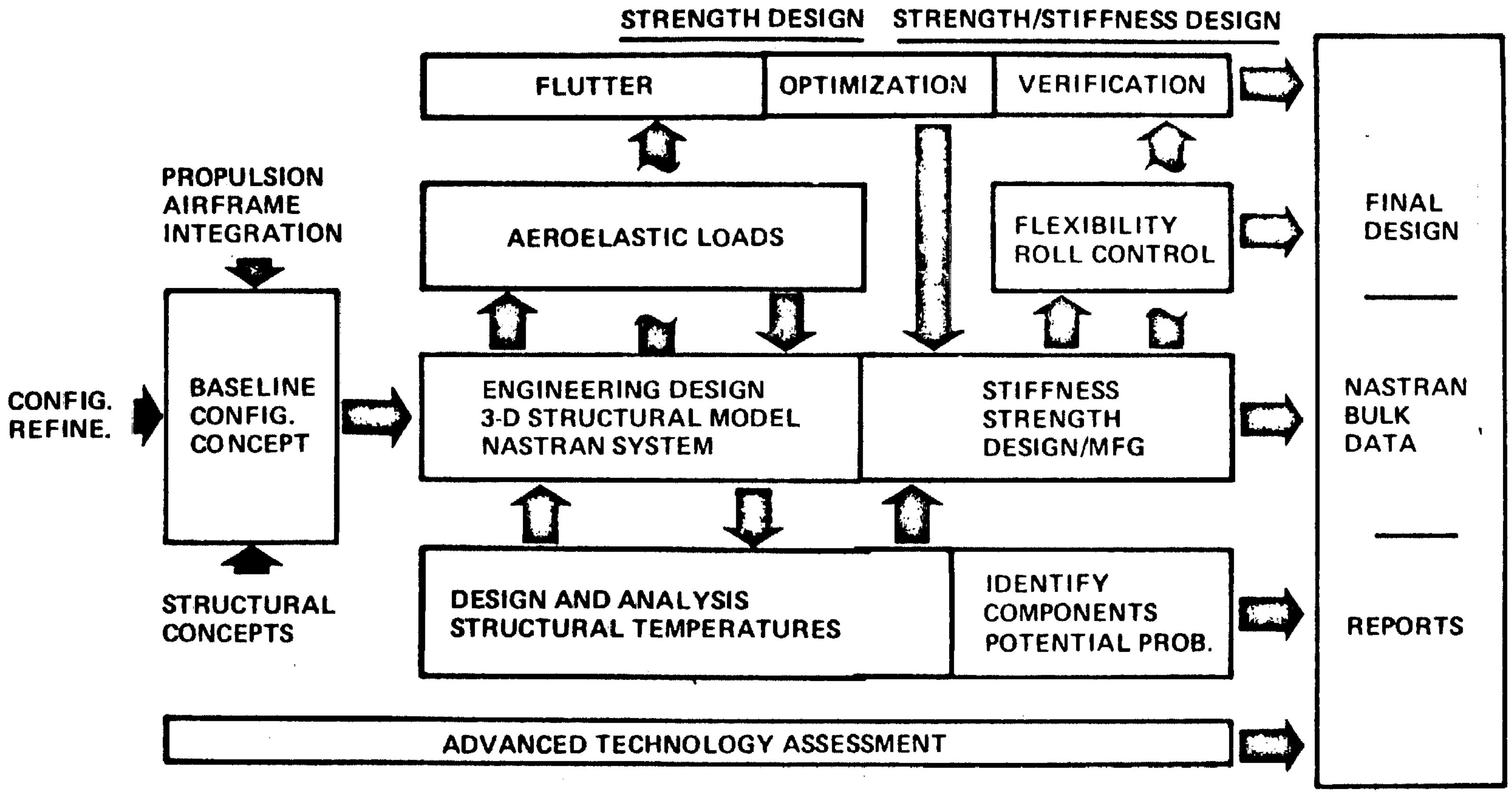
STRENGTH DESIGN
STRENGTH/STIFFNESS DESIGN
FLUTTER
OPTIMIZATION
VERIFICATION
PROPULSION AIRFRAME INTEGRATION
AEROELASTIC LOADS
FLEXIBILITY ROLL CONTROL
FINAL DESIGN
CONFIG. REFINE.
BASELINE CONFIG. CONCEPT
ENGINEERING DESIGN 3-D STRUCTURAL MODEL NASTRAN SYSTEM
STIFFNESS STRENGTH DESIGN/MFG
NASTRAN BULK DATA
STRUCTURAL CONCEPTS
DESIGN AND ANALYSIS STRUCTURAL TEMPERATURES
IDENTIFY COMPONENTS POTENTIAL PROB.
REPORTS
ADVANCED TECHNOLOGY ASSESSMENT

Figure B. Engineering Design and Analyses

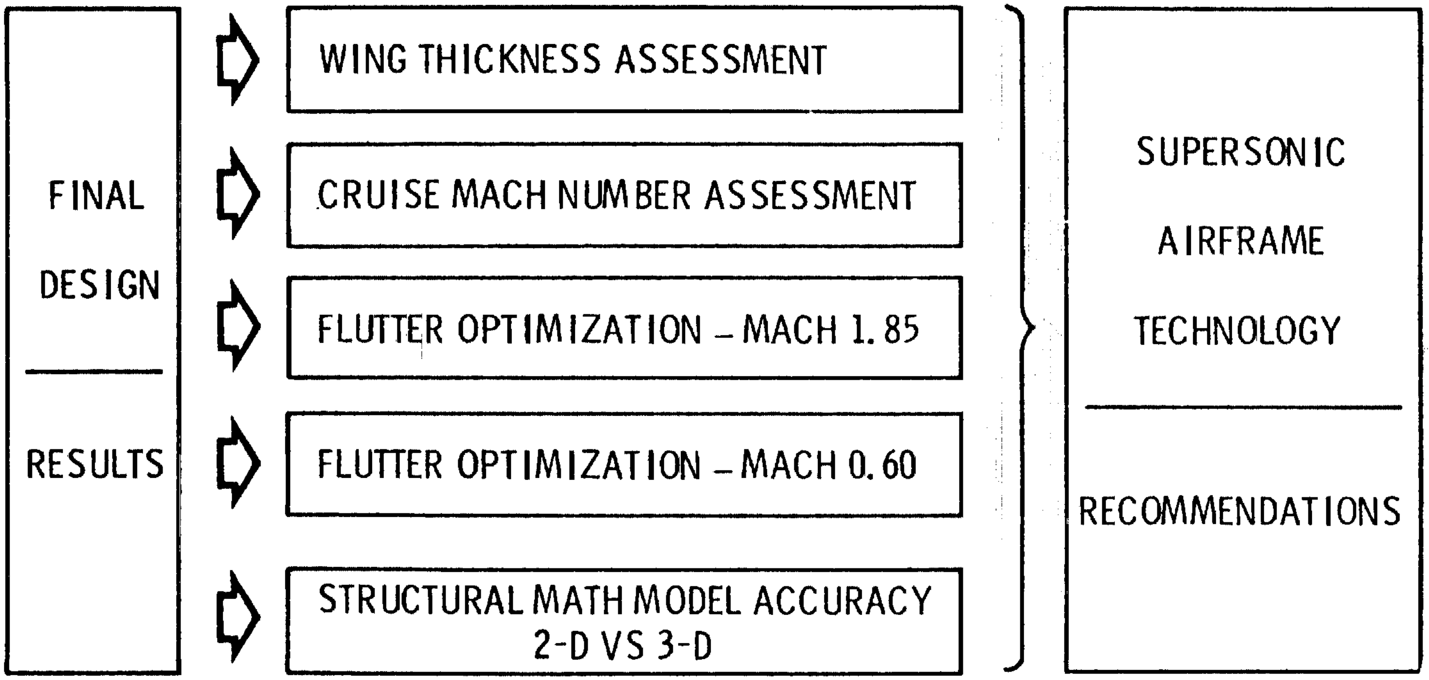
FINAL
DESIGN
RESULTS
WING THICKNESS ASSESSMENT
CRUISE MACH NUMBER ASSESSMENT
FLUTTER OPTIMIZATION – MACH 1.85
FLUTTER OPTIMIZATION – MACH 0.60
STRUCTURAL MATH MODEL ACCURACY 2-D VS 3-D
SUPERSONIC
AIRFRAME
TECHNOLOGY
RECOMMENDATIONS

Figure C. Mass Sensitivity Studies

ACKNOWLEDGEMENT

This investigation was conducted under NASA Contract No. NAS1-12288, Study of Structural Design Concepts for an Arrow-Wing Supersonic Transport Configuration. The study was performed within the Science and Engineering Branch of the Lockheed-California Company, Burbank, California, with the participation of the Lockheed-Georgia Company in a composite design subcontract effort. I. F. Sakata was the Project Engineer, and G.W. Davis was the Lead Engineer. The other contributors to the program are acknowledged at each section.

Mr. J. C. Robinson of the Design Concepts Section, Thermal Structures Branch, and Dr. E. C. Yates, Jr., Computer Aided Methods Branch, Structures and Dynamics Division, NASA Langley Research Center, Hampton, Virginia, were the Technical Representative of the Contracting Officer (TRCO), and Alternate TRCO, respectively for the project.

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SECTION 15

MASS ANALYSIS

by

R.N. Jensen and R.D. Mijares

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SECTION 15

MASS ANALYSIS

INTRODUCTION

The analyses performed to provide structural mass estimates for the arrow-wing supersonic cruise aircraft described in Section 2, are presented in this section.

To realize the full potential for structural mass reduction, a spectrum of approaches for wing and fuselage primary structure design were investigated through analyses, design studies, and detailed design: (1) to assess the relative merits of various structural arrangements, concepts and materials (2) to select the structural approach best suited for the Mach 2.7 environment and (3) to provide construction details and structural mass estimates based on in-depth structural design studies.

BASELINE CONFIGURATION MASS DATA

The interior arrangement of the baseline configuration concepts adopted for the Task I and Task II studies are presented in Figures 15-1 and 15-2, respectively. The dimensional and mass characteristics for the configurations are fully described in Section 2. For completeness, however, the airplane mass property data are presented in the following sections along with a group weight comparison with the study of Reference 1.

Airplane Mass Properties - Task I

Estimated Group Weight and Balance Statement. - An Estimated Group Weight and Balance Statement is presented on Table 15-1 for the Baseline Configuration - Task I. The airplane has a taxi mass of 340,000 kilograms (750,000 pounds), and a range of 7800 kilometers (4200 n. miles) with a payload of 22,000 kilograms (49,000 pounds).

This primarily titanium wing has a total planform area of 1,005 sq. meters and an aspect ratio of 1.62. Its mass includes the center section carry-through structure under the floor, aerodynamic control surfaces and secondary structure. The horizontal stabilizer, and body mounted fin are all-movable. There are also fixed fins outboard on the wing. The body is 90.5 meters long, and will accommodate 234 passengers in five (5) abreast seating. The under floor baggage compartment is located between the nose landing gear and the wing carry-through structure.

The wing mounted main landing gear retracts into a well just outboard of the body. The axisymmetric inlets and duct-burning turbofan engines are under the wing with the thrust reversers just aft of the wing trailing edge. The engines are sized to provide a takeoff thrust to weight ratio of .36.

The mass estimates for the system and equipment reflect composite material application. Standard and operating equipment includes the crew, unusable fuel, and passenger service items.

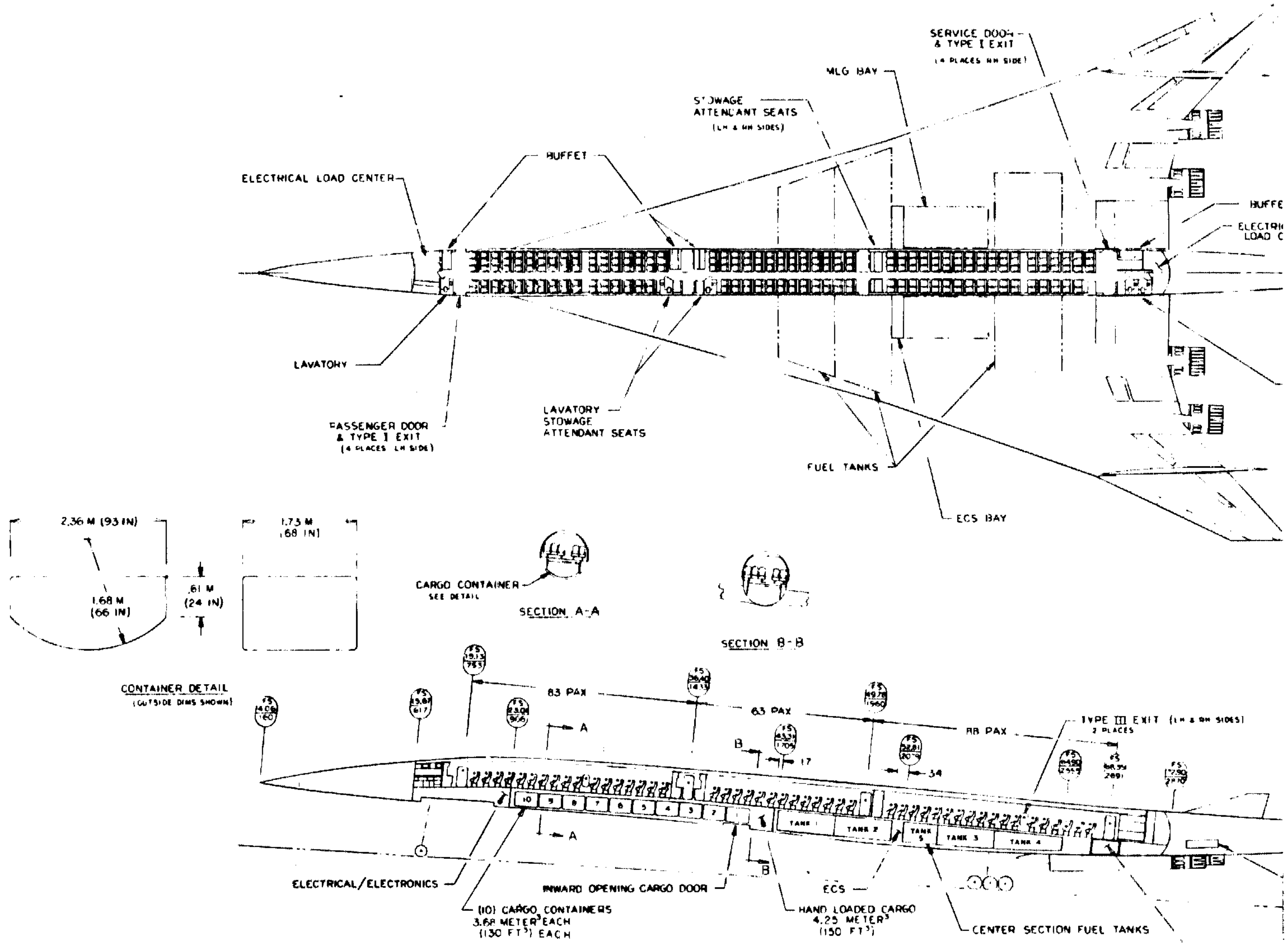
Mass Moment of Inertia. - Airplane mass moments of inertia were determined for the aeroelastic studies. The data for takeoff gross weight, operational weight empty and two intermediate flight conditions are summarized in Table 15-2.

Center of Gravity Travel. - The center of gravity travel is tailored to permit the airplane to cruise with a minimum trim drag penalty. This is accomplished by sequencing the fuel tanks. The forward body and forward wing tanks are used for climbing and accelerating to cruise Mach number. The remaining wing tanks and mid-body tanks are used during cruise. The last two body tanks contain the landing and reserve fuel.

The interior is configured for 234 passengers in five (5) abreast seating with a seat pitch of .86 meters. The baggage is loaded aft of the nose landing gear. Loadability studies indicate unrestricted passenger seating and small curve deviation from the straight payload line. This is primarily due to the low passenger mass to taxi mass fraction.

The fuel tank center of gravities are based on a fuel density of .803 kilogram/liter. The usable fuel volumes are calculated on the basis of 90-percent of the gross contour cross sectional area to allow for structure, systems and usable fuel.

The center of gravity travel shown in Figure 15-3 is used for the Task I Analytical Design Studies. The results of the design, stability and control, and weight and balance studies during Task I are reflected in a new travel diagram for the Engineering Design Study of Task II.


ELECTRICAL LOAD CENTER
BUFFET
STOWAGE
ATTENDANT SEATS
(LH & RH SIDES)
MLG BAY
SERVICE DOOR & TYPE I EXIT
(4 PLACES RH SIDE)
BUFFE
ELECTRI
LOAD C
LAVATORY
PASSENGER DOOR & TYPE I EXIT
(4 PLACES LH SIDE)
LAVATORY
STOWAGE
ATTENDANT SEATS
FUEL TANKS
ECS BAY
2.36 M (93 IN)
1.68 M
(66 IN)
.61 M
(24 IN)
1.73 M
(68 IN)
CONTAINER DETAIL
(OUTSIDE DIMS SHOWN)
CARGO CONTAINER
SEE DETAIL
SECTION A-A
SECTION B-B
FS 4.06 160
FS 15.67 617
FS 19.13 753
FS 23.01 906
83 PAX
FS 36.40 1433
63 PAX
FS 43.31 1705
FS 49.78 1960
88 PAX
FS 52.81 2079
FS 64.90 2555
FS 68.35 2691
FS 72.90 2870
A
A
B
B
17
34
TYPE III EXIT (LH & RH SIDES)
2 PLACES
10
9
8
7
6
5
4
3
2
1
TANK 1
TANK 2
TANK 5
TANK 3
TANK 4
ELECTRICAL/ELECTRONICS
INWARD OPENING CARGO DOOR
(10) CARGO CONTAINERS
3.68 METER³ EACH
(130 FT³) EACH
ECS
HAND LOADED CARGO
4.25 METER³
(150 FT³)
CENTER SECTION FUEL TANKS
HYDRAULIC L

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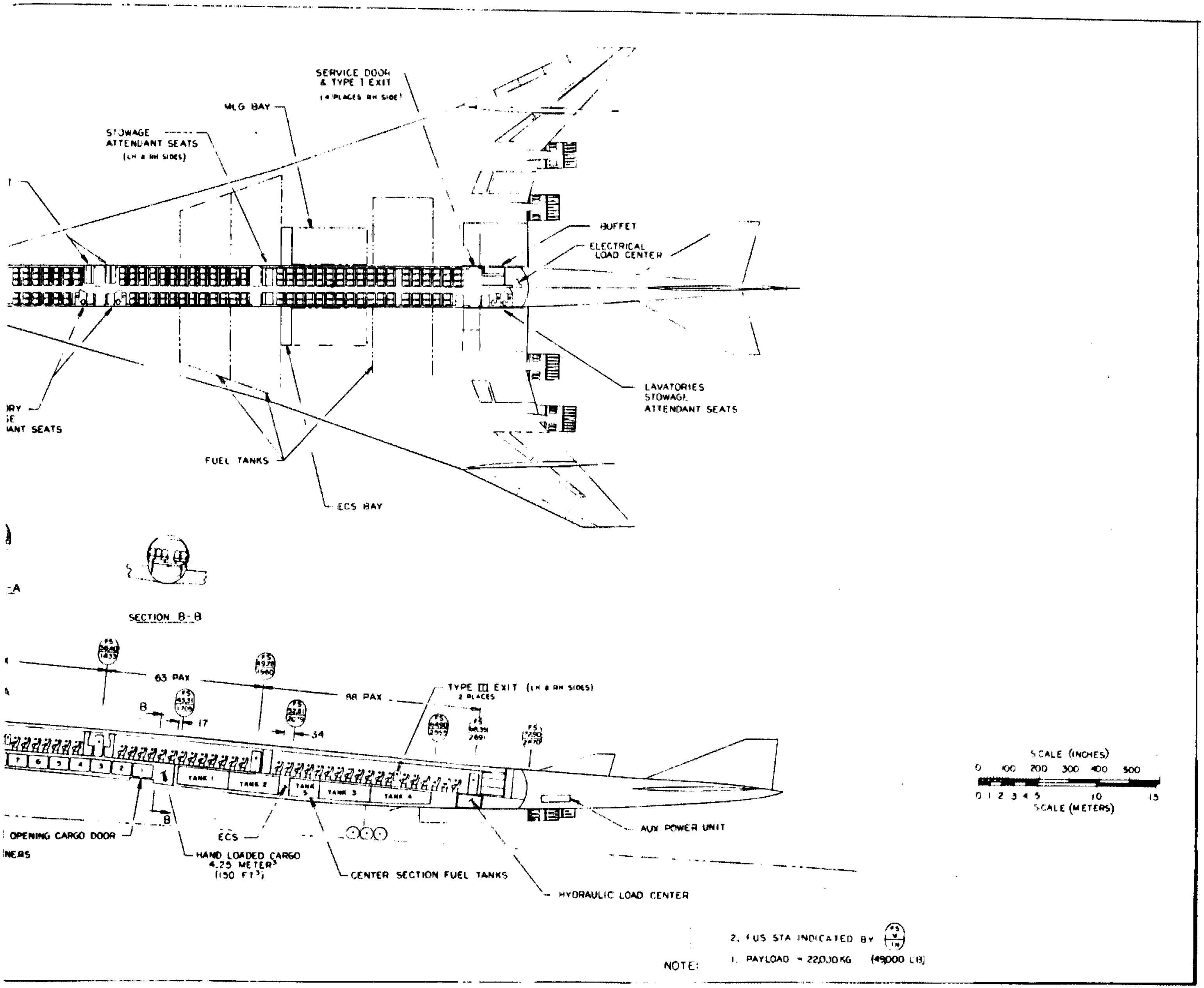
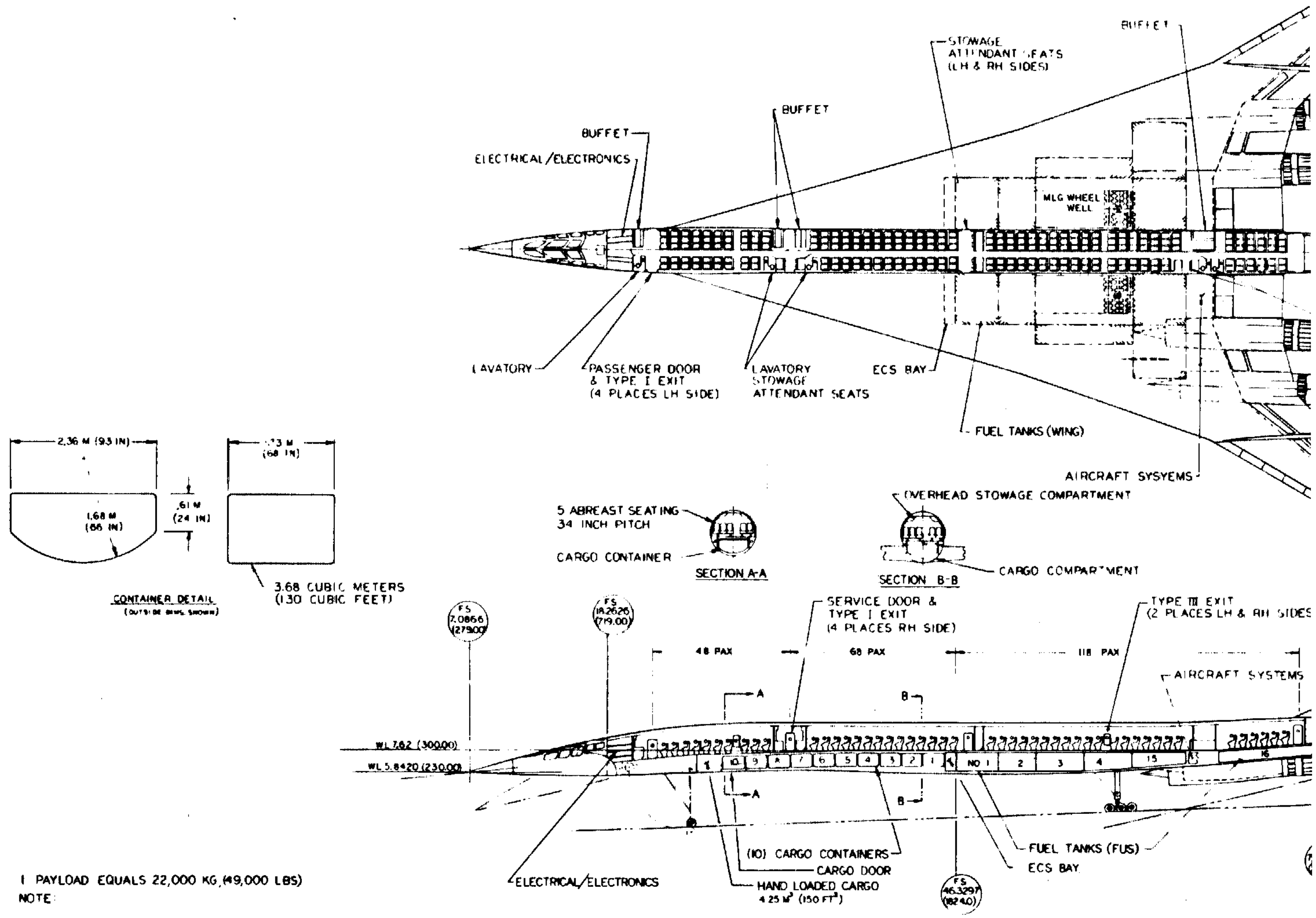

SERVICE DOOR
& TYPE I EXIT
(4 PLACES RH SIDE)
MLG BAY
STOWAGE
ATTENDANT SEATS
(LH & RH SIDES)
BUFFET
ELECTRICAL
LOAD CENTER
LAVATORIES
STOWAGE
ATTENDANT SEATS
FUEL TANKS
ECS BAY
SECTION B-B
63 PAX
88 PAX
B
17
34
TYPE III EXIT (LH & RH SIDES)
2 PLACES
TANK 1
TANK 2
TANK 5
TANK 3
TANK 4
OPENING CARGO DOOR
ECS
HAND LOADED CARGO
4.25 METER³
(150 FT³)
CENTER SECTION FUEL TANKS
AUX POWER UNIT
HYDRAULIC LOAD CENTER
SCALE (INCHES)
0 100 200 300 400 500
0 1 2 3 4 5 10 15
SCALE (METERS)

NOTE:
1. PAYLOAD = 22,000 KG (49,000 LB)
2. FUS STA INDICATED BY (FS / M / IN)

Figure 15-1. Interior Arrangement - Task I


BUFFET
STOWAGE
ATTENDANT SEATS
(LH & RH SIDES)
BUFFET
BUFFET
ELECTRICAL/ELECTRONICS
MLG WHEEL WELL
LAVATORY
PASSENGER DOOR
& TYPE I EXIT
(4 PLACES LH SIDE)
LAVATORY
STOWAGE
ATTENDANT SEATS
ECS BAY
FUEL TANKS (WING)
AIRCRAFT SYSYEMS
2.36 M (93 IN)
1.68 M
(66 IN)
.61 M
(24 IN)
1.73 M
(68 IN)
CONTAINER DETAIL
(OUTSIDE DIMS. SHOWN)
3.68 CUBIC METERS
(130 CUBIC FEET)
5 ABREAST SEATING
34 INCH PITCH
CARGO CONTAINER
SECTION A-A
OVERHEAD STOWAGE COMPARTMENT
CARGO COMPARTMENT
SECTION B-B
F.S.
7.0866
(279.00)
F.S.
18.2626
(719.00)
SERVICE DOOR &
TYPE I EXIT
(4 PLACES RH SIDE)
TYPE III EXIT
(2 PLACES LH & RH SIDES
48 PAX
68 PAX
118 PAX
AIRCRAFT SYSTEMS
A
B
WL 7.62 (300.00)
WL 5.8420 (230.00)
NO 1
2
3
4
15
16
(10) CARGO CONTAINERS
CARGO DOOR
HAND LOADED CARGO
4.25 M³ (150 FT³)
FUEL TANKS (FUS)
ECS BAY
ELECTRICAL/ELECTRONICS
F.S.
46.3297
(1824.0)

NOTE:
1 PAYLOAD EQUALS 22,000 KG (49,000 LBS)

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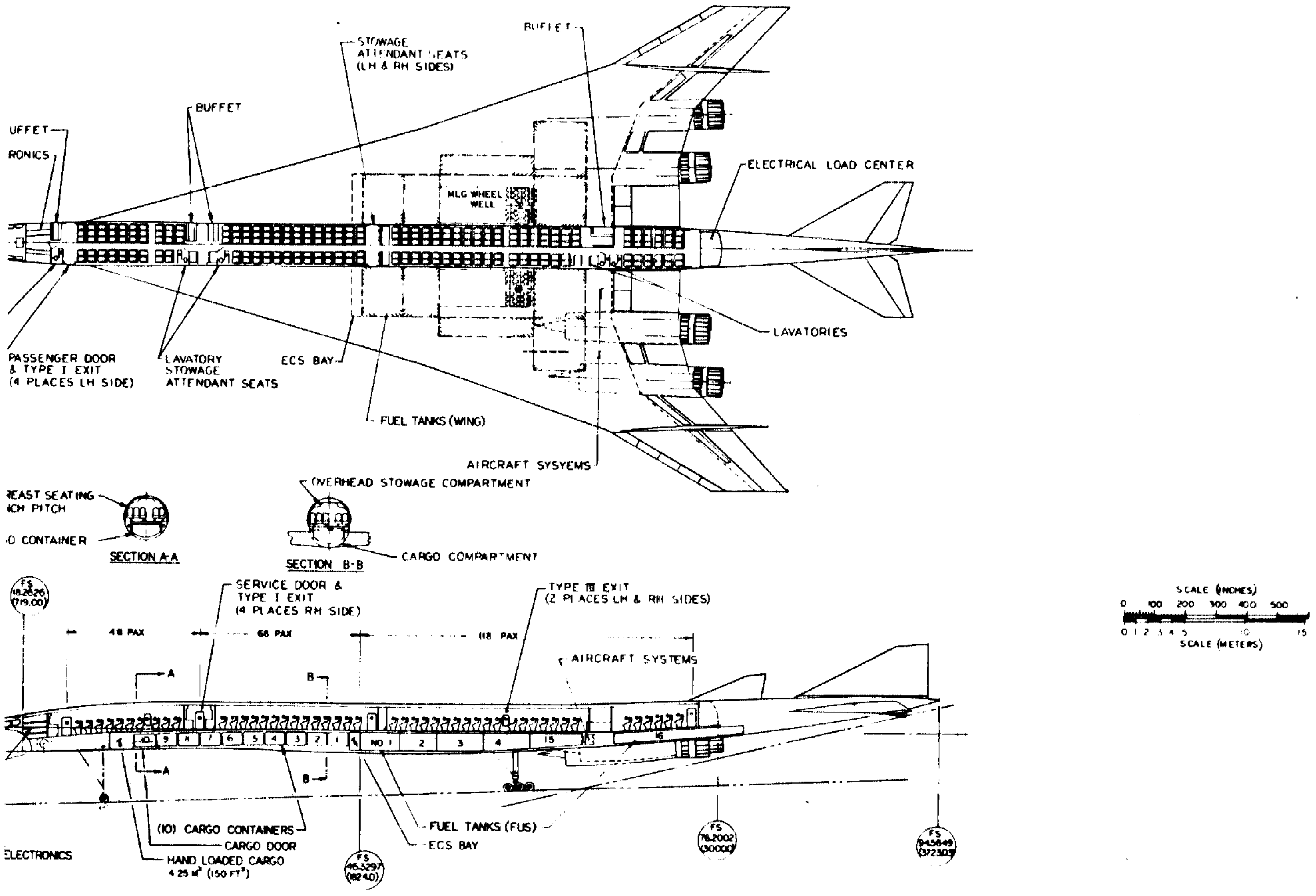

UFFET
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STOWAGE
ATTENDANT SEATS
(LH & RH SIDES)
BUFFET
MLG WHEEL WELL
ELECTRICAL LOAD CENTER
LAVATORIES
PASSENGER DOOR
& TYPE I EXIT
(4 PLACES LH SIDE)
LAVATORY
STOWAGE
ATTENDANT SEATS
ECS BAY
FUEL TANKS (WING)
AIRCRAFT SYSYEMS
REAST SEATING
ICH PITCH
O CONTAINER
SECTION A-A
OVERHEAD STOWAGE COMPARTMENT
SECTION B-B
CARGO COMPARTMENT
FS 18.2626 (719.00)
SERVICE DOOR &
TYPE I EXIT
(4 PLACES RH SIDE)
TYPE III EXIT
(2 PLACES LH & RH SIDES)
48 PAX
68 PAX
118 PAX
AIRCRAFT SYSTEMS
A
B
NO 1
2
3
4
15
16
10
9
8
7
6
5
4
3
2
1
(10) CARGO CONTAINERS
CARGO DOOR
HAND LOADED CARGO
4.25 M³ (150 FT³)
ELECTRONICS
FUEL TANKS (FUS)
ECS BAY
FS 46.3297 (1824.0)
FS 76.2002 (3000.0)
FS 94.5649 (3723.03)
SCALE (INCHES)
0 100 200 300 400 500
0 1 2 3 4 5 10 15
SCALE (METERS)

Figure 15-2. Interior Arrangement - Task II

TABLE 15-1. ESTIMATED GROUP WEIGHT AND BALANCE STATEMENT - TASK I

ITEM	% MAC	MASS (kg)	(lb)	HORIZONTAL X-ARM (m)	(in)
WING		49,713	109,600	58.42	2,300
TAIL – FINS ON WING		1,270	2,800	75.69	2,980
TAIL – FIN ON BODY		1,043	2,300	89.66	3,580
TAIL – HORIZONTAL		2,830	6,240	87.63	3,450
BODY		18,597	41,000	47.55	1,872
LANDING GEAR – NOSE (UP)		1,361	3,000	17.53	690
LANDING GEAR – MAIN (UP)		12,428	27,400	58.32	2,296
AIR INDUCTION		8,074	17,800	67.56	2,660
NACELLES		2,223	4,900	71.55	2,817
PROPULSION – T/F ENGINE INBD.		10,115	22,300	70.87	2,790
PROPULSION – T/F ENGINE OUTBD.		10,115	22,300	72.31	2,847
PROPULSION – SYSTEMS		3,175	7,000	58.42	2,300
SURFACE CONTROLS		3,856	8,500	60.76	2,392
INSTRUMENTS		558	1,230	36.14	1,423
HYDRAULICS		2,585	5,700	58.65	2,309
ELECTRICAL		2,064	4,550	52.10	2,051
AVIONICS		862	1,900	24.64	970
FURNISHINGS & EQUIPMENT		5,216	11,500	43.36	1,707
ECS		3,765	8,300	47.98	1,889
TOLERANCE & OPTIONS		898	1,980	50.80	2,000
MEW	57.7	140,748	310,300	58,98	2,322
STD. & OPER. EQ.		4,887	10,700	43.18	1,700
OEW	56.1	145,635	321,000	58.45	2,201
PAYLOAD		22,000	49,000	42.11	1,658
ZFW	49.9	167,635	370,000	56.29	2,216
FUEL		172,365	380,000	53.01	2,087
TAXI MASS	45.1	340,000	750,000	54.64	2,151

NOSE = 4.064m (160 in.)
MAC = 34.4881 m. (1357.7964 in.) LEMAC X = 39.0855 m. (1538.7961 in.)

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TABLE 15-2. AIRPLANE MASS MOMENT OF INERTIA - TASK I

WEIGHT CONDITION	WEIGHT (LB)	X (IN)	Z (IN)	PITCH	ROLL	YAW
				10^6 SLUG-FT^2		
TAKE OFF GROSS	750,000	2151	–	40.8	6.51	47.3
OPER. EMPTY WT.	321,000	2301	–	27.7	4.68	32.2
INTERMEDIATE I	699,300	2177	−141	39.9	6.36	46.2
● ZERO FUEL	370,000	2216	−128			
● FUEL (A)	329,300	2133	−155			
INTERMEDIATE 2	455,950	2212	−133	35.2	4.75	39.9
● ZERO FUEL	370,000	2216	−128			
● FUEL (B)	85,950	2196	−157			

NOTES: (A) TANKS NOS. 2-5, 8-11 PLUS: 50 PERCENT OF NOS. 1, 6 & 7.
(B) TANKS NOS. 2 & 4 PLUS 50 PERCENT OF NOS. 3 & 5.

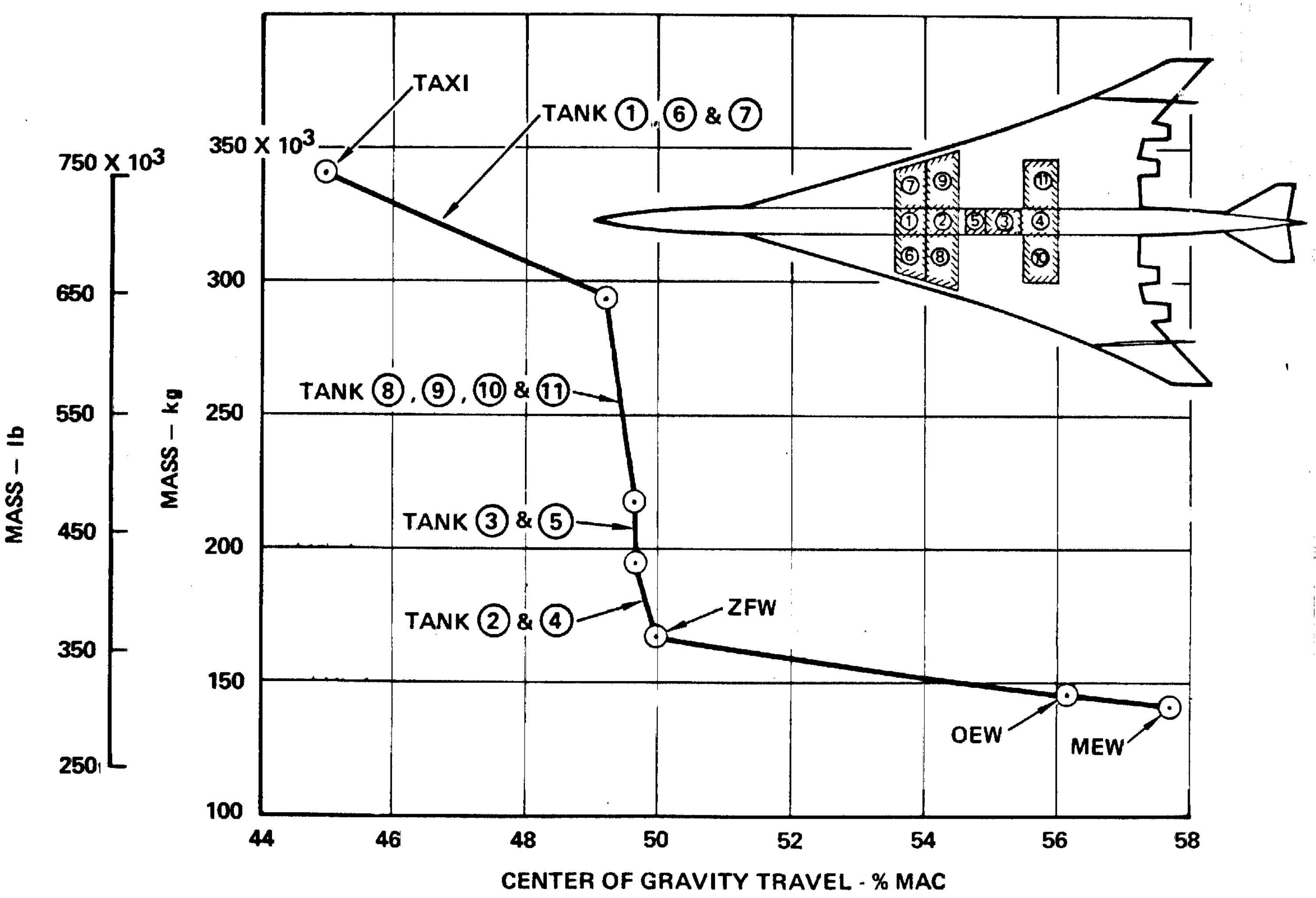

Figure 15-3. Center of Gravity Diagram - Task I

Airplane Mass Properties - Task II

Estimated Group Weight and Balance Statement. - The airplane weight and balance data of Table 15-3 represent the various configurations evaluated during the Task II effort. The data reflect the configuration refinements adopted to the NASA 15F concept. All data are for a fixed sized aircraft with a takeoff gross mass of 340,000 kilograms (750,000 pounds) and payload of 22,000 kilograms (49,000 pounds).

- Task IIA Configuration Data - The Task I weight data (Table 15-1) were adjusted aft to reflect the effect of the configuration changes. The mass of each item was assumed invariant. The taxi mass is at the 52-percent MAC and the zero fuel weight (ZFW) is at the 53.9-percent MAC.
- Task IIB Baseline Data - The data is representative of the configuration changes adopted and the minimum mass wing and fuselage structural approach selected for the Task II effort. The engines have been resized to reflect an uninstalled sea level static thrust of 89,466 pounds per engine and appropriate mass changes for the larger air induction system and nacelles are indicated. The initial mass data does not include allowance for flutter suppression. The taxi mass is 340,000 kilograms (750,000 pounds) with the center of gravity located at the 52.5-percent MAC.
- Task IIB Final Data - The primary mass change is reflected by the increase in wing mass to include the requirements to suppress flutter. A trade off with fuel (Tank No. 16) is made to achieve the same center of gravity location as for the baseline data.

Mass Moment of Inertia. - Airplane mass moment of inertia were computed and plotted in Figure 15-4. The data is similar to that shown in Table 15-2 for the Task I airplane. The pitch moment of inertia is slightly less due to the shortened fuselage while the roll moment of inertia is greater due to the heavier propulsion packages. These data are used for the aeroelastic studies reported in Section 5 and 10.

Center of Gravity Travel. - The fuel management scheduling for airplane center of gravity control is shown in Figure 15-5. The sequencing of fuel is planned (1) to permit the airplane to cruise with a minimum of trim drag penalty and (2) to maximize the heat sink capability of the fuel by emptying the outboard wing tanks as early as possible in the mission.

TABLE 15-3. ESTIMATED GROUP WEIGHT AND BALANCE STATEMENT - TASK II

ITEM	TASK IIA			TASK IIB					
	% MAC	CONFIG. REFINE. DESIGN (LBS)	FUS. STA. (IN.)	% MAC	BASELINE DESIGN (LBS)	STRENGTH DESIGN (LBS)	FUS. STA. (IN.)	FINAL DESIGN (LBS)	FUS. STA. (IN.)
WING		109,600	2,370		89,770	87,660	2,393	90,584	2,409
TAIL – FINS ON WING		2,800	2,991		2,800	2,800	2,991	2,800	2,991
TAIL – FIN ON BODY		2,300	3,522		2,600	2,600	3,522	2,600	3,522
TAIL – HORIZONTAL		6,240	3,449		7,950	7,950	3,449	7,950	3,449
BODY		41,000	1,923		42,100	42,122	1,923	42,122	1,923
LANDING GEAR – NOSE (UP)		3,000	915		3,000	3,000	915	3,000	915
LANDING GEAR – MAIN (UP)		27,400	2,273		27,400	27,400	2,273	27,400	2,273
AIR INDUCTION		17,800	2,667		19,760	19,760	2,626	19,760	2,626
NACELLES		4,900	2,824		5,140	5,137	2,733	5,137	2,783
PROPULSION – T/F ENGINE INBD.		22,300	2,813		25,560	25,562	2,790	25,562	2,790
PROPULSION – T/F ENGINE OUTBD.		22,300	2,870		25,560	25,562	2,834	25,562	2,834
PROPULSION – SYSTEMS		7,000	2,347		7,000	7,007	2,347	7,007	2,347
SURFACE CONTROLS		8,500	2,445		8,500	8,500	2,445	8,500	2,445
INSTRUMENTS		1,230	1,433		1,230	1,230	1,433	1,230	1,483
HYDRAULICS		5,700	2,433		5,700	5,700	2,433	5,700	2,433
ELECTRICAL		4,550	2,132		4,550	4,550	2,132	4,550	2,132
AVIONICS		1,900	1,085		1,900	1,900	1,085	1,900	1,085
FURNISHING & EQUIPMENT		11,500	1,842		11,500	11,500	1,342	11,500	1,842
ECS		8,300	2,009		8,300	8,300	2,009	8,300	2,009
TOLERANCE & OPTIONS		1,980	2,000		1,980	1,980	2,000	1,980	2,000
MEW	60.9	310,300	2,372	62.2	302,300	300,220	2,389	303,144	2,394
STD & OPER. EQ.		10,700	1,819		10,700	10,700	1,819	10,700	1,819
OEW	59.6	321,000	2,353	60.8	313,000	310,920	2,369	313,844	2,374
PAYLOAD		49,000	1,776		49,000	49,000	1,771	49,000	1,771
ZFW	53.9	370,000	2,277	54.8	362,000	359,920	2,288	362,844	2,293
FUEL		380,000	2,226		388,000	390,080	2,228	387,156	2,223
TAXI MASS	52.0	750,000	2,251	52.5	750,000	750,000	2,257	750,000	2,257

LEMAC = FS 1548.2 MAC = 1351.06
X ARM = DISTANCE FROM FUSELAGE
STATION (F.S.) 0
FUS. NOSE AT F.S. 279

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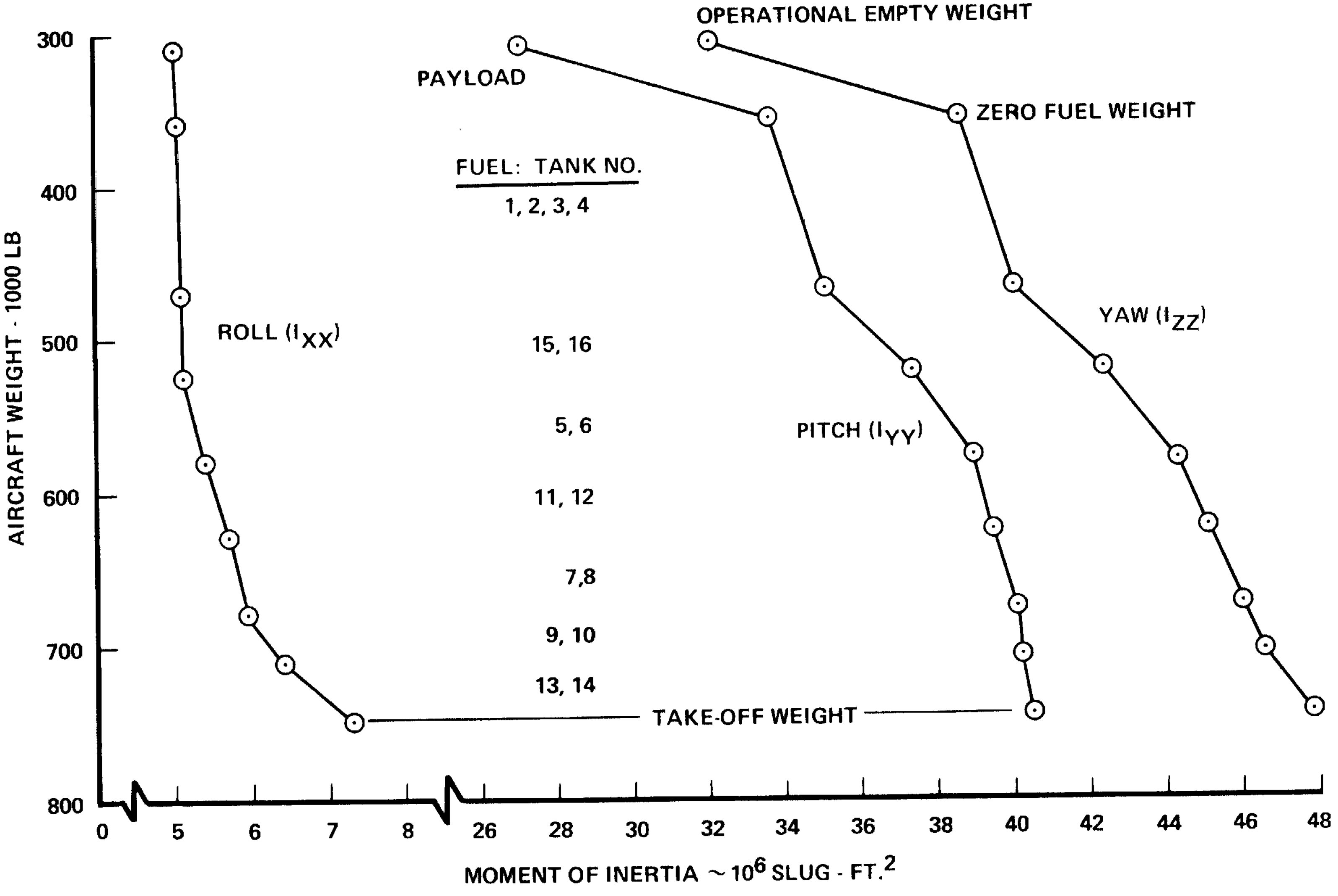

Figure 15-4. Aircraft Moment of Inertia - Task II

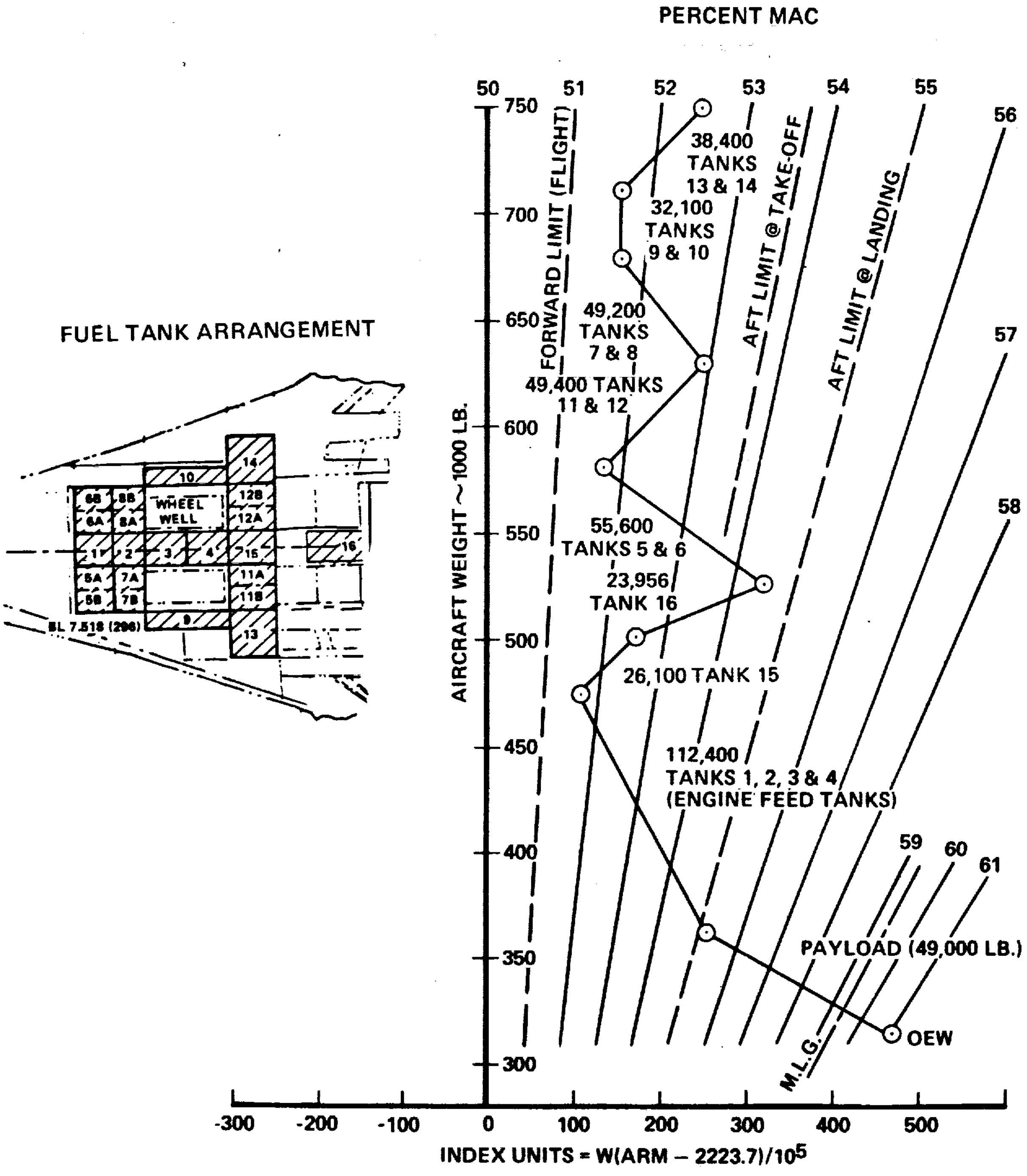

Figure 15-5. Center of Gravity Diagram - Task II

Tanks 1 through 4 are engine feed tanks and are kept full until all other tanks are empty. The usable fuel mass is based on a fuel density of 0.803 kilogram/liter (6.7 pound/gallon) and 90 percent of the gross volume to allow for structure, systems and unusable fuel. The forward limit for flight (51-percent MAC) and the aft limit for takeoff and landing are indicated at 53.5-percent MAC and 55-percent MAC, respectively.

Weight Comparison Data

To compare with previously established weight trends for supersonic cruise aircraft, a group weight comparison was made. The data, presented in Table 15-4, compares a preliminary weight estimate for the arrow-wing supersonic cruise aircraft derived from parametric relationship of the various items (i.e. wing, body) and data of Reference 1. The referenced data is for the Boeing 969-336C production configuration which was obtained by scaling-up-the group weights of the prototype aircraft. As noted on the table, the heavier wing weight used for the starting point of this study is offset by the lighter body structure weight which considers composite application in the cooled interior (i.e. floor beams, post, trim). The larger diameter turbofan engines result in an increase in inlet weights. The equipment and system weight reductions over the referenced data are achieved by utilizing composite materials.

Detail Wing and Body Weights

The scope of the study is to determine the structural approaches best suited for the wing and fuselage design of a Mach 2.7 supersonic cruise aircraft. To identify the relative weights of those components which make up the wing and body structure, these groups were further broken-down into more detailed components as presented in Table 15-5. This procedure isolates different types of structural elements and their relative weights. As the analyses of different elements are completed, the results are compared to the initial estimated values. The relative proportion between the primary structural elements is determined by typical percentages from previous studies and analyses. Other items, such as control surfaces, utilize representative unit weights and their respective areas. Door and windows are based on the size, type, and quantity.

TABLE 15-4. PRELIMINARY GROUP WEIGHT COMPARISON

REFERENCE	LOCKHEED ARROW-WING INITIAL DATA	BOEING 969-336C PRODUCTION
	PARAMETRIC DATA	REFERENCE 1
ITEM	WEIGHT (lbs)	WEIGHT (lbs)
WING	109,600	92,700
TAIL – HORIZONTAL	4,400	2,370
– VERTICAL	3,800	3,270
– CANARD	–	2,950
BODY	41,000	51,570
LAND. GEAR – NOSE	3,000	3,030
MAIN	27,400	27,910
AIR INDUCTION	17,200	15,650
NACELLE	6.800	
TOTAL STRUCTURE	(213,200)	(199,450)
PROPULSION – ENGINES	46,000	45,020
– SYSTEMS	7,000	6,310
SURFACE CONTROLS	8,500	12,450
INSTRUMENTS	1,230	3,400
HYDRAULICS	5,700	5,600
ELECTRICAL	4,550	5,050
AVIONICS	1,900	2,690
FURN. AND EQUIPMENT	11,500	21,290
ECS	8,300	8,100
OPTIONS AND TOLERANCES	2,420	5,480
MANUF. EMPTY WT. (MEW)	310,300	314,840
STD AND OPER EQ.	10,700	11,810
OPER. EMPTY WT. (OEW)	321,000	326,650
PAYLOAD	49,000	48,906
ZERO FUEL WEIGHT (ZFW)	370,000	375,556
FUEL	380,000	374,444
TAXI WEIGHT	750,000	750,000

TABLE 15-5. ESTIMATED WEIGHTS FOR WING AND BODY - TASK I

ITEM		WEIGHT (lbs)	
		COMPONENT	GROUP
WING GROUP:			109,600
CENTER SECTION		17,000	
SURFACE MATERIAL	12,750 lb.		
SHEAR MATERIAL	3,400		
RIBS	850		
OUTER PANEL		66,620	
SURFACE MATERIAL	49,960		
SHEAR MATERIAL	7,330		
RIBS	9,330		
LEADING EDGE		5,470	
TRAILING EDGE		5,520	
MLG DOORS		3,600	
BODY FAIRING		1,600	
AILERONS		1,440	
T.E. FLAPS		6,880	
L.E. FLAPS		1,220	
SPOILERS		250	
BODY GROUP:			41,000
BULKHEADS AND FRAMES		4,940	
SKINS		10,510	
LONGERONS AND STIFFENERS		6,010	
NOSE AND FLIGHT STATION		2,500	
NLG WELL		900	
WINDSHIELD AND WINDOWS		1,680	
FLOORING AND SUPPORTS		3,820	
DOORS AND MECHANISM		4,170	
UNDERWING FAIRING		1,870	
CARGO COMPARTMENT PROVISIONS		1,060	
WING-BODY FITTINGS		1,500	
TAIL-BODY FITTINGS		600	
PROVISIONS FOR SYSTEMS		740	
FINISH AND SEALING		700	

STRUCTURAL MODEL MASS DATA

Grid Point Distribution - Task I

The data from the Estimated Group Weight and Balance Statement of Table 15-1 are distributed to the structural model grid points (SIC) for use in the static loads and flutter analysis programs. For the initial effort, a single mass distribution which is representative of the three structural arrangements is used.

Table 15-6 presents the Operating Weight Empty (OWE), Table 15-7 the payload, and Table 15-8 the fuel distribution by tank. Several individual components lumped in the SIC distribution are listed separately in Table 15-9.

The negative sense of weight values at some points are the result of applying a couple to obtain the correct center of gravity for the overhanging vertical fins. The mass moment of inertia for the propulsion system, wing, and fuel are presented in Tables 15-10 and 15-11. These data are based on the weight distribution at the SIC grid points. The grid point locations are defined in Section 9, Structural Analysis Models.

Grid Point Distribution - Task IIA

For the Task IIA investigation, the Task I weight data (Table 15-1) were adjusted aft to reflect the configuration changes adopted. The configuration refinements are shown in Figure 15-6. The major configuration differences are delineated below:

(1) Added wing area (50 sq. ft./side) outboard of BL 470 by reducing angle of the leading edge from 64.64-degrees to 60-degrees.

(2) Increased number of fuel tanks and changed the tank arrangement to achieve an aft shift in center-of-gravity.

(3) Reduced length of fuselage forebody by 119 inches, payload moved aft.

(4) Increased fuselage-mounted vertical tail area from 290 sq. feet to 325 sq. feet.

The geometric parameters for these changes are defined in Table 15-12.

The appropriate changes to reflect the aforementioned refinements were made (data not included) including changes to the wing tip surface panel distribution to reflect strength-designed thicknesses (Figure 15-7). These data were then input to the static loads and flutter analysis programs.

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TABLE 15-6. OPERATING WEIGHT EMPTY DISTRIBUTION - TASK I

SIC PT.	GP NO.	X IN.	Y IN.	W *
1	0102	200	0	280
2	0104	400	↕	1284
3	0106	600	↕	2300
4	0108	800	↕	3157
5	0110	1000	↕	3789
6	3151	1210	0	2644
26	0212	1210	62	450
7	3251	1382	0	2807
27	0214	1382	62	496
43	0314	1382	125	793
8	3351	1580	0	2561
28	0216	↕	62	495
44	0316	↕	125	1043
59	0416	1580	196	500
9	3451	1680	0	1778
29	0218	↕	62	485
45	0318	↕	125	672
60	0418	↕	196	544
74	0518	1680	232	454
10	3551	1772	0	1994
30	0220	↕	62	491
46	0320	↕	125	647
61	0420	↕	196	531
75	0520	↕	232	450
88	0620	1772	266	70
11	3651	1865	0	1984
31	0222	↕	62	485
47	0322	↕	125	640
62	0422	↕	196	520
76	0522	↕	232	486
89	0722	1865	296	655
12	3751	1955	0	2222
32	0224	↕	62	485
48	0324	↕	125	625
63	0424	↕	196	520
77	0524	↕	232	478
90	0724	↕	296	655
101	0824	1955	335	70
13	3851	2045	0	2346
33	0226	↕	62	2394
49	0326	↕	125	659
64	0426	↕	196	2432
78	0526	↕	232	461
91	0726	↕	296	655
102	0926	2045	365	403
14	3951	2145	0	2368
34	0228	↕	62	454
50	0328	↕	125	659
65	0428	↕	196	500
79	0528	↕	232	457
92	0728	↕	296	653
103	0928	↕	365	400
112	1028	2145	406	368
15	4051	2235	0	2294
35	0230	↕	62	450
51	0330	↕	125	647
66	0430	↕	196	490
80	0530	↕	232	439
93	0730	↕	296	655
104	0930	↕	365	400
113	1030	↕	406	368
121	1130	2235	435	60
16	4151	2330	0	2247
36	0232	↕	62	5373
52	0332	↕	125	594
67	0432	↕	196	5406
81	0532	↕	232	439
94	0732	↕	296	646
105	0932	↕	365	400
114	1032	↕	406	367
122	1232	2330	470	537
17	4251	2405	0	1520
37	0234	↕	62	445
53	0334	↕	125	542
68	0434	↕	196	481
82	0534	↕	232	428
95	0734	2405	296	604
106	0934	2410	365	400
115	1034	2415	406	369
123	1234	2420	470	536
18	4351	2485	0	1793
38	0236	↕	62	429
54	0336	↕	125	560
69	0436	↕	196	480
83	0536	↕	232	1185
96	0736	2485	296	1354
107	0936	2500	365	360
116	1036	2508	406	1032
124	1236	2520	470	1110
19	4451	2565	0	2887
39	0238	↕	62	435
55	0338	↕	125	542
70	0438	↕	196	480
84	0538	↕	232	944
97	0738	2565	296	1122

* pounds/side

TABLE 15-6. OPERATING WEIGHT EMPTY DISTRIBUTION - TASK I (Continued)

SIC PT.	GP NO.	X IN.	Y IN.	W *
108	0938	2590	365	360
117	1038	2603	406	848
125	1238	2625	470	1027
20	4551	2640	0	876
40	0240	↕	62	420
56	0340	↕	125	508
71	0440	↕	196	469
85	0540	↕	232	896
98	0740	2640	296	1084
109	0940	2678	365	360
118	1040	2700	406	810
126	1240	2730	470	952
41	0242	2710	62	424
57	0342	↕	125	750
72	0442	↕	196	456
86	0542	2710	232	845
99	0742	2713	296	1035
110	0942	2743	365	363
119	1042	2763	406	761
127	1242	2798	470	903
21	0143	2800	0	4114
42	0246	2855	62	107
58	0346	↕	125	215
73	0446	↕	196	215
87	0546	↕	232	107
100	0746	2855	296	75
111	0946	2868	365	150
120	1046	2880	406	75
128	1246	2900	470	109
22	0145	3000	0	2172
23	0146	3200	0	1020
24	0147	3360	0	1822
25	0148	3470	0	4448

SIC PT.	GP NO	X IN.	Y IN.	W *
129	1300	2400	496	162
130	1304	2478	524	164
131	1310	2562	554	162
133	1312	2595	510	124
132	1320	2645	585	143
134	1322	2670	552	124
135	1324	2698	515	121
136	1326	2696	603	172
137	1328	2717	576	170
138	1330	2742	543	171
139	1332	2774	502	143
140	1346	2772	640	205
141	1348	2790	616	-284
142	1350	2812	588	-284
143	1352	2839	552	127
144	1354	2860	523	328
145	1522	2849	676	157
146	1524	2863	656	156
147	1526	2882	634	140
148	1528	2904	603	2802
149	1530	2924	578	163
150	1534	2959	534	218
151	1540	3001	578	109
152	1562	2922	711	116
153	1564	2934	695	117
154	1566	2949	676	116
155	1568	2968	652	117
156	1570	2986	630	116
158	1610	3024	664	42
157	1614	3050	632	50
159	1622	2996	746	115
160	1624	3006	734	115
161	1626	3017	720	116

SIC PT.	GP NO.	X IN.	Y IN.	W *
162	1628	3032	701	52
163	1630	3046	682	52
164	1634	3067	650	52
165	1662	3047	771	74
166	1664	3055	761	75
167	1666	3064	750	75
168	1668	3075	735	25
169	1670	3088	718	25
170	1674	3105	695	26
171	1702	3098	794.4	75
172	1704	3104	787	75
173	1706	3111	779	75
174	1708	3119	769	25
175	1710	3131	754	25
176	1714	3145	737	25
177	1724	3116	794.4	56
178	1746	3134	794.4	56
179	1768	3152	794.4	41
180	1790	3177	794.4	15
182	1794	3183	780	15
181	1798	3197	794.4	17
183	0660	2660	264	3795
184	0662	2800	264	9795
185	1160	2720	438	3684
186	1162	2855	438	9382

* Pounds/side

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TABLE 15-7. PAYLOAD DISTRIBUTION - TASK I

SIC PT.	GP NO.	X (y=0)	W Lbs/Side
4	0108	800	1240
5	0110	1000	2847
6	3151	1210	3680
7	3251	1382	2440
8	3351	1580	2440
9	3451	1680	1440
10	3551	1772	1240
11	3651	1865	1240
12	3751	1955	620
13	3851	2045	620
14	3951	2145	1240
15	4051	2235	1240
16	4151	2330	1240
17	4251	2405	1000
18	4351	2485	1000
19	4451	2565	973

TABLE 15-8. FUEL DISTRIBUTION - TASK I

SIC PT.	GP NO.	X IN.	Y IN.	FUEL TANK No.						
				1	6	2	8	5	3	TOTAL (LB/SIDE)
9	3451	1680	0	1050	-					1050
29	0218		62	1050	800					1850
45	0318		125		1500					1500
60	0418		196		1500					1500
74	0518	1680	232		200					200
10	3551	1772	0	4200	-					4200
30	0220		62	4200	3600					7800
46	0320		125		6600					6600
61	0420		196		5000					5000
75	0520		232		2400					2400
88	0620	1772	266		1200					1200
11	3651	1865	0	2600	-	1650	-			4250
31	0222		62	2600	2200	1650	1600			8050
47	0322		125		4000		3000			7000
62	0422		196		2800		2100			4900
76	0522		232		1900		1400			3300
89	0722	1865	296		1300		900			2200
12	3751	1955	0			4250	-			4250
32	0224		62			4250	3900			8150
48	0324		125				6900			6900
63	0424		196				4500			4500
77	0524		232				4200			4200
90	0724		296				3100			3100
101	0824	1955	335				900			900
13	3851	2045	0			2175	-	725		2900
33	0226		62			2175	1900	725		4800
49	0326		125				3300			3300
64	0426		196				2300			2300
78	0526		232				2200			2200
91	0726		296				1800			1800
102	0926	2045	365				1000			1000
14	3951	2145	0					4100	700	4800
34	0228	2145	62					4100	700	4800
	TOTAL (LB/SIDE)			15700	35000	16150	45000	9650	1400 (Partial)	122900

TABLE 15-8. FUEL DISTRIBUTION - TASK I (Continued)

SIC PT.	GP NO.	X IN.	Y IN.	FUEL TANK NO. 3	FUEL TANK NO. 4	FUEL TANK NO. 10	TOTAL (LB/SIDE)
15	4051	2235	0	4000			4000
35	0230	2235	62	4000			4000
16	4151	2330	0	2800	1000		3800
36	0232	2330	62	2800	1000		3800
17	4251	2405	0		2600	-	2600
37	0234	↑	62		2600	2900	5500
53	0334		125			5400	5400
68	0434		196			3800	3800
82	0534	↓	232			2700	2700
95	0734	2405	296			2600	2600
18	4351	2485	0		2650	-	2650
38	0236	↑	62		2650	3000	5650
54	0336		125			5500	5500
69	0436		196			4200	4200
83	0536	↓	232			2800	2800
96	0736	2485	296			2700	2700
19	4451	2565	0		1000	-	1000
39	0238	↑	62		1000	560	1560
55	0338		125			1010	1010
70	0438		196			780	780
84	0538	↓	232			530	530
97	0738	2565	296			520	520
TOTAL (LB/SIDE)				13600 (Partial)	14500	39000	67100

TABLE 15-9. INDIVIDUAL COMPONENT WEIGHT DISTRIBUTION - TASK I

SIC POINT	GRID POINT NO.	X (in.)	Y (in.)	WEIGHT lb/SIDE	ITEM
24	0147	3360	0	567	HORIZONTAL TAIL
25	0148	3470	0	2553	HORIZONTAL TAIL
24	0147	3360	0	-469	VERTICAL TAIL (MOVABLE)
25	0148	3470	0	+469 +860	VERTICAL TAIL (MOVABLE)
33	0226	2045	62	1923	MLG (UP)
36	0232	2330	62	4928	MLG (UP)
64	0426	2045	196	1922	MLG (UP)
67	0432	2330	196	4927	MLG (UP)
3	0106	600	0	525	NLG (UP)
4	0108	800	0	975	NLG (UP)
141	1348	2790	616	-510	VERTICAL TAIL – WING
142	1350	2812	588	-510	VERTICAL TAIL – WING
148	1528	2904	603	+1020 +1400	VERTICAL TAIL – WING
183	0660	2660	264	3795	ENGINES, NAC. AND PROP. SYS.
184	0662	2800	264	9795	ENGINES, NAC. AND PROP. SYS.
185	1160	2720	438	3684	ENGINES, NAC. AND PROP. SYS.
186	1162	2855	438	9382	ENINGES, NAC. AND PROP. SYS.

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TABLE 15-10. MOMENT OF INERTIA OF ENGINE, NACELLE, AND PROPULSION SYSTEM - TASK I

SIC POINT	GRID PT. NO.	WEIGHT (lbs/SIDE)	$\bar{X}$ (in.)	$\bar{Y}$ (in.)	$\bar{Z}$ (in.)	1000 SLUG - ft^2	
						I_{o_X}	$I_{o_Y} = I_{o_Z}$
183	0660	13,590	2761	264	-215	3.39	21.70
184	0662						
185	1160	13,066	2817	438	-207	3.23	20.83
186	1162						

TABLE 15-11. MOMENT OF INERTIA OF WING STRUCTURE AND FUEL - TASK I

ITEM		WEIGHT (lbs/SIDE)	$\bar{X}$ (in.)	$\bar{Y}$ (in.)	10^6 SLUG - ft^2
					I_{o_Z}
WING STRUCTURE		75,737	2322	252	2.69
FUEL		154,500	2082	145	2.71
Σ	TOTAL	230,237	2161	181	6.15

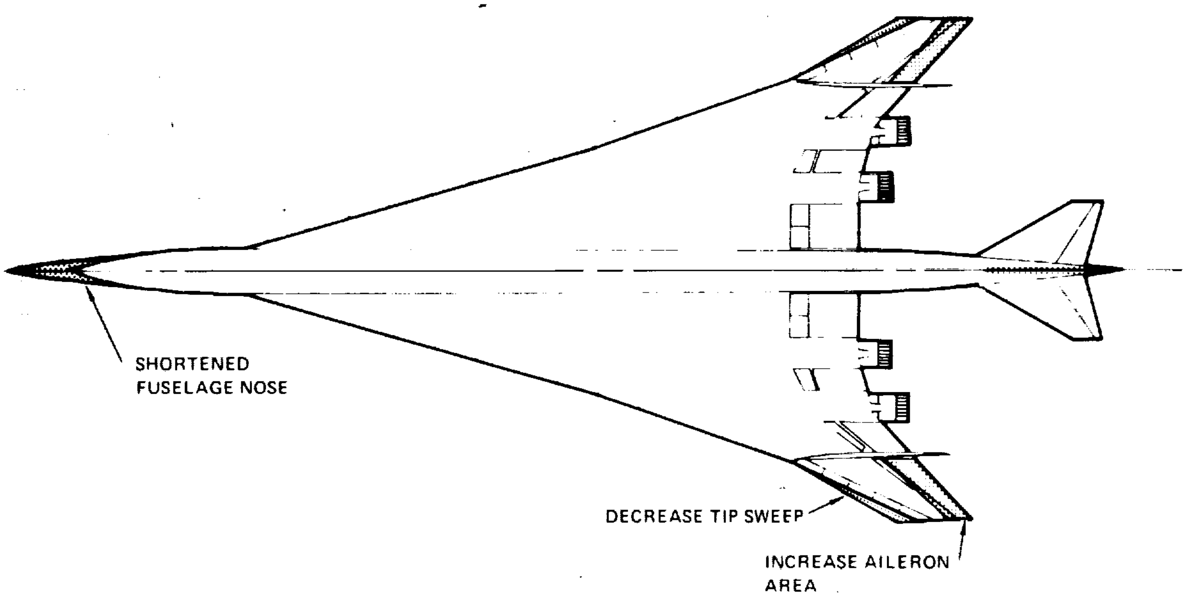
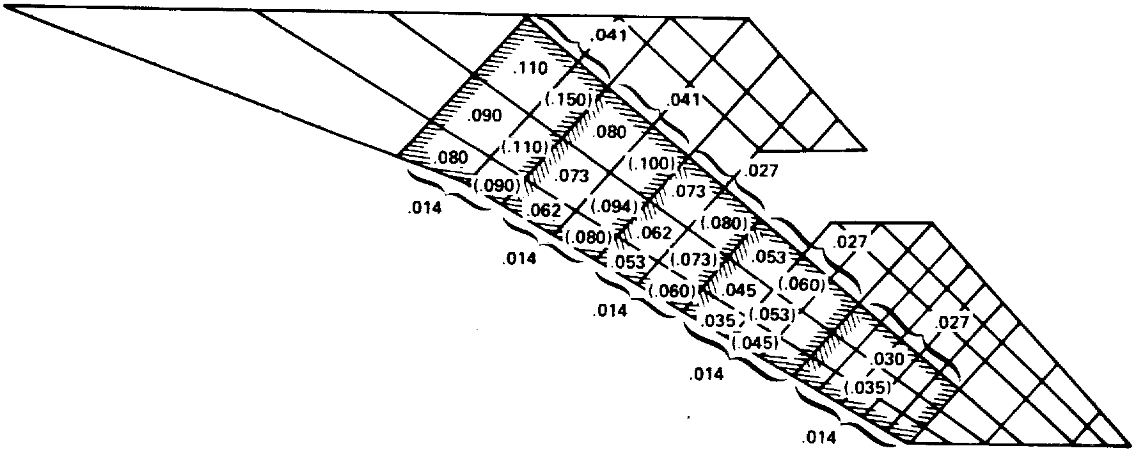

SHORTENED
FUSELAGE NOSE
DECREASE TIP SWEEP
INCREASE AILERON
AREA

Figure 15-6. Configuration Comparison - Task I and Task II


.041
.110
.090
(.150)
.041
.080
(.110)
.080
(.100)
.027
.073
.073
(.090)
.014
.062
(.094)
(.080)
(.080)
.062
.027
.014
.053
(.073)
.053
(.060)
(.060)
.045
.014
.035
(.053)
.027
(.045)
.030
.014
(.035)
.014

NOTE:

XXX = UPPER SURFACE EFFECTIVE THICKNESS (IN)
(XXX) = LOWER SURFACE EFFECTIVE THICKNESS (IN)
⁓ = BEAM WEB THICKNESS (IN)

Figure 15-7. Surface Panel Thickness - Strength Design

TABLE 15-12. AIRPLANE GEOMETRIC PARAMETERS - TASK I AND TASK II

			TASK I	TASK II
WING:	AREA (PARALLEL TO Z_0 PLANE)	ft^2	10,822	10,923
	AR		1.62	1.607
	λ		0.08	0.113
	b	in.	1590	1590
	Cr	in.	2195.5	2195.5
	C_t	in.	175.6	249.2
	MAC	in.	1357.8	1351.3
	L.E. SWEEP to BL 392	(degree)	74	74
	to BL 602	(degree)	70.84	70.84
	to BL 795	(degree)	64.64	60.00
CONTROL SURFACES:	L.E. FLAP AREA	(ft^2)	159	133
	SPOILERS – PLAIN	(ft^2)	120	110
	SPOILER – SLOT – DEFLECTORS	(ft^2)	120	115
	FLAPS – INBOARD	(ft^2)	316	306
	– FLAPERONS/AILERONS	(ft^2)	310/180	247/250
HORIZONTAL TAIL:	AREA (WL PLANE)	(ft^2)	795	795
	AR		1.607	1.607
	λ		0.225	0.225
	b	(in.)	441.6	441.6
	ELEVATOR AREA (2)	(ft^2)	174	174
FUSELAGE VERTICAL TAIL:	AREA (MOVEABLE)	(ft^2)	290	325
	AR		0.517	0.517
	λ		0.23	0.23
	b	(in.)	146.4	155.5
WING VERTICAL TAIL:	AREA (FIXED - 2)	(ft^2)	466	466
	AR		0.495	0.495
	λ		0.136	0.136
	b	(in.)	129.0	129.0
FUSELAGE:	LENGTH	(in.)	3564.0	3444.0
	WIDTH	(in.)	135.0	135.0
	DEPTH	(in.)	166.0	166.0

Grid Point Distribution - Task IIB (Strength/Stiffness)

The configuration refinements identified in Section 2 and summarized in the Grid Point Distrubition discussions were adopted for the Task IIB strength and strength/stiffness design effort. The structural approach selected for these analyses is a Hybrid structural arrangement consisting of the chordwise stiffened design for the wing structure inboard of BL 406 and the monocoque design for the wing tip structure.

As shown in Table 15-3 the Task IIA weight distribution was updated to include the engine size and weight increases to properly reflect the uninstalled sea level static thrust level of 89,466 pounds per engine instead of the 77,957 pounds per engine. Appropriate nacelle and air induction system weights were also included.

Table 15-13 presents the Operating Empty Weight (OEW) distribution for the strength/stiffness design; the payload and fuel (tanks 1 through 16) distributions are detailed in Table 15-14. The concentrated weight items which are included in the OEW distribution of Table 15-13 are identified separately in Table 15-15. The weight, center of gravity and moment of inertia data for tail surfaces and engines are contained in Table 15-16. These data were applied to the flutter analysis effort reported in Section 10.

The final strength/stiffness distribution resulting from the flutter optimization analysis is shown in Table 15-17 and pictorially displayed on Figure 15-8. To provide adequate torsional stiffness, a total weight increment of 1462 pounds per side is added to the wing tip box structure. To maintain constant aircraft gross weight, an equal weight is removed from Tank 16 fuel, as shown in Table 15-18.

The mass moment of inertia for the wing and contents is presented in Table 15-19. The data includes the wing fuel (BL 62 - BL 406) and payload distribution to BL 62.0. These data are based on the weight distribution at the SIC grid points. The moment of inertia data for the aileron and outboard flaperon are shown in Table 15-20. These data are based on an expression for calculation of the mass moment of inertia derived from the L-1011 wide body transport.

TABLE 15-13. OPERATING WEIGHT EMPTY DISTRIBUTION - TASK IIB STRENGTH/STIFFNESS DESIGN

MODEL GRID ID	WEIGHT (lb/SIDE)	COORDINATES X (in.)	COORDINATES Y (in.)
3150	785	400.00	00.00
3250	2,235	600.00	00.00
3350	2,967	800.00	00.00
3450	2,893	1,000.00	00.00
3550 •	1,282	1,100.00	00.00
0112	1,040	1,210.00	00.00
0114	1,395	1,382.00	00.00
0116	733	1,580.00	00.00
0118	509	1,680.00	00.00
0120	508	1,772.00	0.000
0122	517	1,865.00	00.00
0124	432	1,955.00	00.00
0126	453	2,045.00	00.00
0128	462	2,145.00	00.00
0130	450	2,235.00	00.00
0132	454	2,330.00	00.00
0134	411	2,405.00	00.00
0136	427	2,485.00	00.00
0138	1,092	2,565.00	00.00
0140	1,143	2,640 00	00.00
5150	2,512	2,800.00	00.00
5250	2,702	3,000.00	00.00
5350	952	3,200.00	00.00
5450	1,027	3,360.00	00.00
5550	5,902	3,470.00	00.00
SUBTOTAL	(33,283)	(2,109.40)	
0314	710	1,382.00	125.00
0316	345	1,580.00	125.00
0318	225	1,680.00	125.00
0320	265	1,772.00	125.00
0322	336	1,865.00	125.00
0324	331	1,955.00	125.00
0326	442	2,045.00	125.00
0328	1,114	2,145.00	125.00
0330	8,068	2,235.00	125.00
0212	2,923	1,210.00	62.00
0214	3,036	1,382.00	62.00
0216	1,654	1,580.00	62.00
0218	1,154	1,680.00	62.00
0220	1,330	1,772.00	62.00
0222	1,309	1,865.00	62.00
0224	1,136	1,955.00	62.00
0226	1,186	2,045.00	62.00
0228	1,815	2,145.00	62.00
0230	1,785	2,235.00	62.00
0232	1,686	2,330.00	62.00
0234	1,164	2,405.00	62.00
0236	1,163	2,485.00	62.00
0238	2,586	2,565.00	62.00
0240	2,637	2,640.00	62.00
0242	288	2,710.00	62.00
0246	100	2,855.00	62.00
SUBTOTAL	(26,952)	(1,999.50)	
0416	480	1,580.00	196.00
0418	175	1,680.00	196.00
0420	220	1,772.00	196.00
0422	287	1,865.00	196.00
0424	282	1,955.00	196.00
0426	392	2,045.00	196.00
0428	310	2,145.00	196.00
0430	1,614	2,235.00	196.00
0432	1,126	2,330.00	196.00
0434	370	2,405.00	196.00
0436	335	2,485.00	196.00
0438	430	2,565.00	196.00
0440	600	2,640.00	196.00
0442	768	2,710.00	196.00
0446	155	2,855.00	196.00
SUBTOTAL	(7,544)	(2,274.60)	

REFER TO SECTION 9 STRUCTURAL ANALYSIS MODELS FOR GRID POINT LOCATIONS ON AIRCRAFT PLANFORM

TABLE 15-13. OPERATING WEIGHT EMPTY DISTRIBUTION - TASK IIB STRENGTH/STIFFNESS DESIGN (Continued)

MODEL GRID ID	WEIGHT	COORDINATES	
	(lb/SIDE)	X (in.)	Y (in.)
0332	6,180	2,330.00	125.00
0334	420	2,405.00	125.00
0336	400	2,485.00	125.00
0338	510	2,565.00	125.00
0340	700	2,640.00	125.00
0342	900	2,710.00	125.00
0346	200	2,855.00	125.00
SUBTOTAL	(21,146)	(2,248.30)	
0518	255	1,680.00	232.00
0520	115	1,772.00	232.00
0522	176	1,865.00	232.00
0524	170	1,955.00	232.00
0526	232	2,045.00	232.00
0528	282	2,145.00	232.00
0530	276	2,235.00	232.00
0532	280	2,330.00	232.00
0534	355	2,405.00	232.00
0536	325	2,485.00	232.00
0538	1,655	2,565.00	232.00
0540	1,610	2,640.00	232.00
0542	1,035	2,710.00	232.00
0546	140	2,855.00	232.00
SUBTOTAL	(6,906)	(2,462.00)	
0926	242	2,045.00	365.00
0928	198	2,145.00	365.00
0930	187	2,235.00	365.00
0932	192	2,330.00	365.00
0934	390	2,410.00	365.00
0936	365	2,500.00	365.00
0938	330	2,590.00	365.00
0940	560	2,678.00	365.00
0942	756	2,743.00	365.00
0946	145	2,868.00	365.00
SUBTOTAL	(3,365)	(2,520.40)	
0620	215	1,772.00	266.00
0722	275	1,865.00	299.50
0724	182	1,955.00	296.00
0726	242	2,045.00	296.00
0728	342	2,145.00	296.00
0730	336	2,235.00	296.00
0732	340	2,330.00	296.00
0734	400	2,405.00	296.00
0736	400	2,485.00	296.00
0738	1,745	2,565.00	296.00
0740	1,695	2,640.00	296.00
0742	1,100	2,713.00	296.00
0746	200	2,855.00	296.00
0824	187	1,955.00	332.50
SUBTOTAL	(7,659)	(2,461.00)	
1130	149	2,235.00	435.00
1232	192	2,330.00	470.00
1234	230	2,420.00	470.00
1236	405	2,520.00	470.00
1238	2,355	2,625.00	470.00
1240	1,060	2,730.00	470.00
1242	1,010	2,798.00	470.00
1246	140	2,900.00	470.00
SUBTOTAL	(5,541)	(2,646.70)	
1522	230	2,818.00	676.00
1524	154	2,831.50	660.00
1526	202	2,854.00	633.00
1528	2,569	2,882.00	600.00
1530	450	2,905.50	573.00
1534	125	2,949.00	521.50

REFER TO SECTION 9 STRUCTURAL ANALYSIS MODELS FOR GRID POINT LOCATIONS ON AIRCRAFT PLANFORM

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TABLE 15-13. OPERATING WEIGHT EMPTY DISTRIBUTION - TASK IIB STRENGTH/STIFFNESS DESIGN (Continued)

MODEL GRID ID	WEIGHT	COORDINATES	
	(lb/SIDE)	X (in.)	Y (in.)
1028	160	2,145.00	402.00
1030	121	2,235.00	406.00
1032	186	2,330.00	406.00
1034	245	2,415.00	406.00
1036	280	2,508.00	406.00
1038	1,585	2,603.00	406.00
1040	1,605	2,700.00	406.00
1042	1,020	2,763.00	406.00
1,046	130	2,880.00	406.00
SUBTOTAL	(5,332)	(2,624.30)	
1300	132	2,399.00	495.00
1304	181	2,475.50	523.00
1310	168	2,555.50	552.00
1312	275	2,589.50	511.70
1320	131	2,636.50	581.30
1322	180	2,659.20	554.30
1324	263	2,691.00	516.80
1326	-488	2,686.80	600.00
1328	173	2,703.00	581.00
1330	238	2,732.80	546.50
1332	477	2,769.00	503.30
1346	254	2,755.70	639.50
1348	197	2,770.00	622.00
1350	318	2,796.00	592.00
1352	426	2,828.50	554.00
1354	536	2,854.00	524.00
SUBTOTAL	(3,461)	(2,726.00)	
1622	192	2,961.00	758.00
1624	106	2,971.80	745.20
1626	129	2,987.00	727.20
1628	214	3,005.50	705.50

MODEL GRID ID	WEIGHT	COORDINATES	
	(lb/SIDE)	X (in.)	Y (in.)
1540	180	2,998.00	573.00
1562	245	2,882.00	712.00
1564	165	2,894.00	698.00
1566	222	2,914.00	675.30
1568	304	2,937.70	647.00
1570	340	2,958.50	623.00
1610	250	3,007.00	668.00
1614	75	3,046.00	623.00
SUBTOTAL	(5,511)	(2,901.50)	
1724	13	3,054.10	795.00
1746	20	3,082.90	795.00
1768	26	3,111.30	795.00
1790	104	3,141.30	795.00
1794	95	3,174.50	756.00
1798	25	3,211.00	795.00
SUBTOTAL	(283)	(3,147.30)	

REFER TO SECTION 9 STRUCTURAL ANALYSIS MODELS FOR GRID POINT LOCATIONS ON AIRCRAFT PLANFORM

TABLE 15-13. OPERATING WEIGHT EMPTY DISTRIBUTION - TASK IIB
STRENGTH/STIFFNESS DESIGN (Continued)

MODEL GRID ID	WEIGHT	COORDINATES	
	(lb/SIDE)	X (in.)	Y (in.)
1630	122	3,023.80	684.00
1634	45	3,062.00	639.00
1662	69	2,993.00	777.00
1664	53	3,003.00	764.20
1666	78	3,017.00	748.30
1668	92	3,033.50	729.00
1670	140	3,050.30	709.00
1674	55	3,087.50	665.50
1702	26	3,025.10	795.00
1704	39	3,035.00	783.00
1706	65	3,046.50	769.00
1708	104	3,061.00	752.00
1710	190	3,077.00	734.00
1714	90	3,113.00	691.50
SUBTOTAL	(1,809)	(3,025.80)	
ENGINE & NACELLE:			
0660	1,306	2,660.00	264.00
0662	12,759	2,800.00	264.00
SUBTOTAL	(14,065)	(2,787.00)	
ENGINE & NACELLE:			
1160	2,500	2,720.00	438.00
1162	11,565	2,855.00	438.00
SUBTOTAL	(14,065)	(2,831.00)	
WING FIN:	(REFERENCE)		
1326	-781		
1528	2,181		
SUBTOTAL	(1,400)	(2,990.90)	
TOTAL O.E.W.	156,922	2,374.40	–

REFER TO SECTION 9 STRUCTURAL ANALYSIS MODELS FOR GRID POINT LOCATIONS ON AIRCRAFT PLANFORM

TABLE 15-14. PAYLOAD AND FUEL DISTRIBUTION - TASK II STRENGTH/STIFFNESS DESIGN (LB/SIDE)

MODEL GRID	X (in.)	PAY-LOAD	TANK NUMBER ρ FUEL = 6.7 lb/gal										
			1	2	3	4	15	16	5	7	9	11	13
Y = 0													
					TOTAL FUEL CAPACITY/SIDE = 196,800 lb @ F.S. 2233.7								
3450	1000	2,084											
3550	1100	2,068											
0112	1210	914											
0114	1382	1,380											
0116	1580	1,489											
0118	1680	974											
0120	1772	342											
0122	1865	62	5,110										
0124	1955	557	2,190	1,778									
0126	2045	588		4,622									
0128	2145	587			6,400								
0130	2235	569			800	1,667							
0132	2330	526				5,532							
0134	2405	482					2,352						
0136	2485	464					3,072						
0138	2565	–					1,100						
0140	2640	218						1,500					
5150	2800	2,038						6,800					
5250	3000	–						5,400					
		(15,342)											
Y = 62.0 in.													
0212	1210	914											
0214	1382	1,380											
0216	1580	1,490											
0218	1680	975											
0220	1772	343											
0222	1865	62	5,110						3,660				
0224	1955	557	2,190	1,778					1,570	1,290			
0226	2045	588		4,622						3,340			
0228	2145	588			6,400								
0230	2235	569			800	1,668							
0232	2330	526				5,533							
0234	2405	483					2,353					1,670	
0236	2435	464					3,073					2,190	
0238	2565	–					1,100					795	
0240	2640	219						1,500					
		(9,158)											

REFER TO SECTION 9 STRUCTURAL ANALYSIS MODELS FOR GRID POINT LOCATIONS ON AIRCRAFT PLANFORM

TABLE 15-14. PAYLOAD AND FUEL DISTRIBUTION - TASK II STRENGTH/STIFFNESS DESIGN (LB/SIDE) (Continued)

MODEL GRID	X (in.)	PAY-LOAD	TANK NUMBER ρ FUEL = 6.7 lb/gal										
			1	2	3	4	15	16	5	7	9	11	13
Y = 135.0 in.													
0322	1865								7,680				
0324	1955								3,290	2,700			
0326	2045									7,100			
0334	2405											3,490	
0336	2485											4,580	
0338	2565											1,665	
Y = 196.0 in.													
0422	1865								6,060				
0424	1955								2,600	2,130			
0426	2045									5,440			
0434	2405											2,760	
0436	2485											3,625	
0438	2565											1,315	
Y = 232.0 in.													
0522	1865								2,060				
0524	1955								880	720			
0526	2045									1,880			
0528	2145										2,875		
0530	2235										2,675		
0532	2330										2,475		
0534	2405											935	1,455
0536	2485											1,230	1,765
0538	2565											445	420
Y = 296.0 in.													
0728	2145										2,875		
0730	2235										2,675		
0732	2330										2,475		
0734	2405												3,040
0736	2485												3,695
0738	2565												875
Y = 365.0 in.													
0934	2410												2,450
0936	2500												2,980
0933	2590												405
Y = 406.0 in.													
1034	2415												955
1036	2508												1,160
TOTALS/SIDE		24,500	14,600	12,800	14,400	14,400	13,050	15,200	27,800	24,600	16,050	24,700	19,200
$\bar{X}$ =		1771.3	1892.0	2020.0	2155.0	2308.0	2469.6	2839.5	1892.0	2020.0	2232.0	2470.0	2465.5

REFER TO SECTION 9 STRUCTURAL ANALYSIS MODELS FOR GRID POINT LOCATIONS ON AIRCRAFT PLANFORM

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TABLE 15-15. INDIVIDUAL COMPONENT WEIGHT DISTRIBUTION - TASK IIB STRENGTH/STIFFNESS DESIGN

ITEM	GRID I.D.	WEIGHT (lb/SIDE)	$\overline{X}$ (in.)	$\overline{Y}$ (in.)	$\overline{Z}$ (in.)
NOSE LANDING GEAR (UP)		1,500	914.7	0	269.10
	3350	640	800.0	0	262.64
	3450	860	1000.0	0	273.97
MAIN LANDING GEAR (UP)		13,700	2273.0	137.2	297.80
	0330	6,810	2235.0	125.0	296.00
	0430	1,410	2235.0	196.0	304.00
	0332	4,540	2330.0	125.0	297.00
	0432	940	2330.0	196.0	306.00
VERTICAL TAIL – FUS. (MOVABLE)		1,300	3522.0	0	377.30
	5450	-615	3360.0	0	358.30
	5550	+1,915	3470.0	0	371.20
HORIZONTAL TAIL – FUS. (MOVABLE)		3,975	3449.0	0	368.70
	5450	760	3360.0	0	358.30
	5550	3,215	3470.0	0	371.20
AIR INDUCTION – INBOARD		4,940	2602.5	264.0	316.20
	0538	1,235	2565.0	232.0	313.00
	0738	1,235	2565.0	296.0	319.00
	0540	1,235	2640.0	232.0	313.00
	0740	1,235	2640.0	296.0	320.00
AIR INDUCTION – OUTBOARD		4,940	2649.7	488.0	322.10
	1033	1,235	2603.0	406.0	321.00
	1238	1,930	2625.0	470.0	319.50
	1040	1,235	2700.0	406.0	326.00
	1240	540	2730.0	470.0	325.20
ENGINES AND NACELLES – INBOARD		14,065	2787.0	264.0	305.00
	0660	1,306	2660.0	264.0	305.00
	0662	12,759	2800.0	264.0	305.00
ENGINES AND NACELLES – OUTBOARD		14,065	2831.0	438.0	311.00
	1160	2,500	2720.0	438.0	311.00
	1162	11,565	2855.0	438.0	311.00
WING FIN		1,400	2990.9	600.0	338.20
	1326	-781	2686.3	600.0	307.80
	1528	+2,181	2882.0	600.0	327.30

REFER TO SECTION 9 STRUCTURAL ANALYSIS MODELS FOR GRID POINT LOCATIONS ON AIRCRAFT PLANFORM

TABLE 15-16. MASS DATA FOR FLUTTER ANALYSIS - TASK IIB

ITEM	WEIGHT (lb/SIDE)	CENTER OF GRAVITY inch			MOM. OF INERTIA – ~ Slug-ft^2		
		$\overline{X}$	$\overline{Y}$	$\overline{Z}$	I_{X_o}	I_{Y_o}	I_{Z_o}
VERTICAL TAIL ~ FUS.	1,300	3522.0	0	482.0	455	3,158	2,703
HORIZONTAL TAIL ~ FUS.	3,975	3449.0	75.3	375.0	2,091	6,159	8,250
ENGINES AND NACELLES:							
INBOARD:	14,065	2787.0	264.0	258.0	3,795	17,709	17,709
OUTBOARD:	14,065	2831.0	438.0	272.5	3,795	17,709	17,709
FIN ~WING	1,400	2990.9	600.0	383.0	542	3,134	2,592
SECT. W.L.							
1 312-340	240	2920	600	330	1.7	293.5	291.8
2 340-370	374	2952	600	354	6.1	611.5	605.4
3 370-400	332	2993	600	384	5.4	507.6	502.2
4 400-430	229	3034	600	414	3.7	203.1	199.4
5 430-455	125	3067	600	441	1.4	59.0	57.6
6 455-475	63	3096	600	464	0.5	14.2	13.7
7 475-TIP	37	3122	600	484	0.2	3.9	3.7

NOTES:

REFER TO SECTION 9 STRUCTURAL ANALYSIS MODELS FOR GRID POINT LOCATIONS OF AIRCRAFT PLANFORM.

CENTER OF GRAVITY DIFFERENCES BETWEEN TABLE 15-16 AND TABLE 15-15 ARE DUE TO 2-D GRID POINT LOCATIONS IN TABLE 15-15 NOT COINCIDENT WITH ACTUAL CENTER OF GRAVITY.

TABLE 15-17. FINAL MASS DISTRIBUTION - STRENGTH VS STRENGTH/STIFFNESS DESIGN

MODEL GRID ID	WEIGHT (lbs)		MODEL GRID ID	WEIGHT (lbs)	
	STRENGTH ONLY	STRENGTH/ STIFFNESS		STRENGTH ONLY	STRENGTH/ STIFFNESS
1300	132	132	1610	250	250
1304	168	181	1614	75	75
1310	156	168	1622	164	192
1312	270	275	1624	50	106
1320	150	131	1626	60	129
1322	180	180	1628	146	214
1324	240	263	1630	122	122
1326	-504	-488	1634	45	45
1328	144	173	1662	54	69
1330	190	238	1664	22	53
1332	410	477	1666	33	78
1346	234	254	1668	46	92
1348	138	197	1670	140	140
1350	257	318	1674	55	55
1352	364	426			
1354	536	536	1702	16	26
			1704	19	39
1522	210	230	1706	25	65
1524	103	154	1708	44	104
1526	131	202	1710	190	190
1528	2493	2569	1714	90	90
1530	450	450	1724	8	13
1534	125	125	1746	15	20
1540	180	180	1768	22	26
1562	208	245	1790	104	104
1564	84	165	1794	95	95
1566	108	222	1798	25	25
1568	190	304			
1570	340	340			
			TOTALS	9,602	11,064

REFER TO SECTION 9 STRUCTURAL ANALYSIS MODELS (FIGURE 9-5)

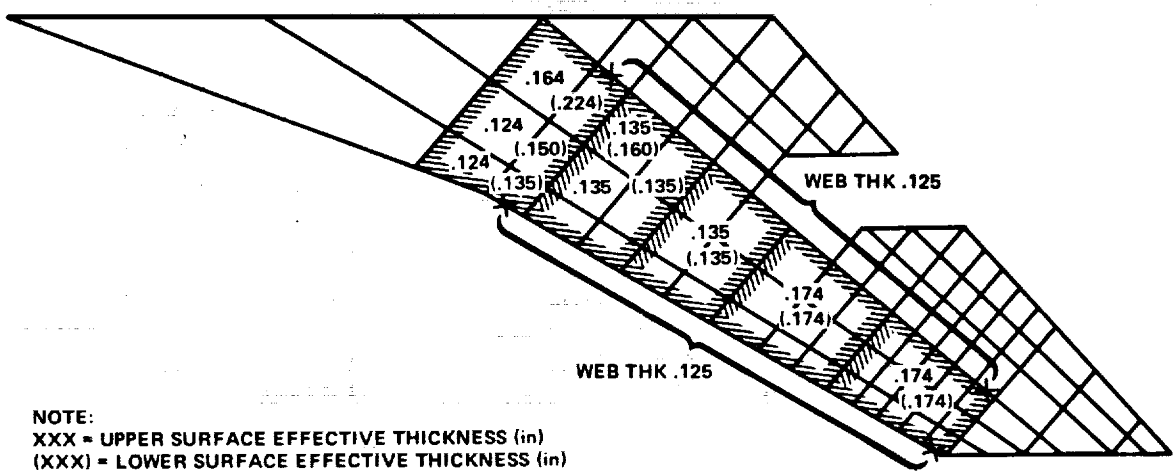

.164
(.224)
.124
(.150)
.124
(.135)
.135
(.160)
.135
(.135)
WEB THK .125
.135
(.135)
.174
(.174)
WEB THK .125
.174
(.174)
NOTE:
XXX = UPPER SURFACE EFFECTIVE THICKNESS (in)
(XXX) = LOWER SURFACE EFFECTIVE THICKNESS (in)

Figure 15-8. Surface Panel Thickness - Final Design

TABLE 15-18. FUEL DISTRIBUTION FOR TANK NO. 16 - STRENGTH VS STRENGTH/STIFFNESS DESIGN

MODEL GRID ID	TANK 16 FUEL (lb/side)	
	STRENGTH ONLY	STRENGTH/ STIFFNESS
0140	1,326	1,182
5150	6,013	5,359
5250	4,775	4,255
0240	1,326	1,182
TOTALS	13,440	11,978
REF. TANK 16 FUEL CAPACITY = 15,200 lb/SIDE		

TABLE 15-19. MOMENT OF INERTIA - WING, PAYLOAD AND FUEL - TASK II

ITEM	WEIGHT (lb/SIDE)	$\overline{X}$ (F.S.)	$\overline{Y}$ (B.L.)	$\overline{Z}$ (W.L.)	$I_{ZZ_{c.g.}}$ 10^6 Slug - ft^2
WING AND CONTENTS @ OEW (A) (BL 62 TO TIP)	92,647	2320.1	223.1	275	4.32
PAYLOAD (BL.62)	9,158	1820.4	62.0	310	0.34
FUEL (BL 62 to BL 406)	148,303	2191.0	160.8	270	2.18
WING AND CONTENTS @ TOW	250,108	2225.3	180.3	273.3	7.46

(A) DOES NOT INCLUDE WING FIN AND ENGINES

TABLE 15-20. MOMENT OF INERTIA - AILERON AND OUTBOARD FLAPERON

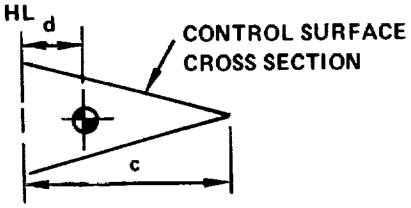
CONTROL SURFACE	AVERAGE (A) CHORD, C (in.)	WEIGHT, W (lb)	I_{HL} (B)	
			(lb-in.^{2})	(lb-in.-sec^2)
AILERON	65	625	0.602×10^6	1560
OUTBOARD FLAPERON	76	504	0.662×10^6	1720

(A) MEASURED NORMAL TO HINGE LINE

(B) $I_{HL} = \left[\left(\frac{Wc^2}{12}\right) + (Wd^2)\right]$

$I_{HL} = 0.228\,Wc^2$ (lb-in.^{2}) WITH $d = 0.38c$

$I_{HL} = .00059\,Wc^2$ (lb-in.-sec^2)



STRUCTURAL CONCEPT MASS ANALYSIS

Method

The Structural Concept Analysis section presents sized elements for selected wing and fuselage locations for each structural arrangement. The data reflects variable spar or rib spacing for each panel concept at the point design region specified. To determine the unit weights between analysis areas, the load/temperature map is utilized. This permits unit weights to be increased or decreased with corresponding changes in the load and temperature environment. Consideration is also given to the lightly loaded minimum gage regions.

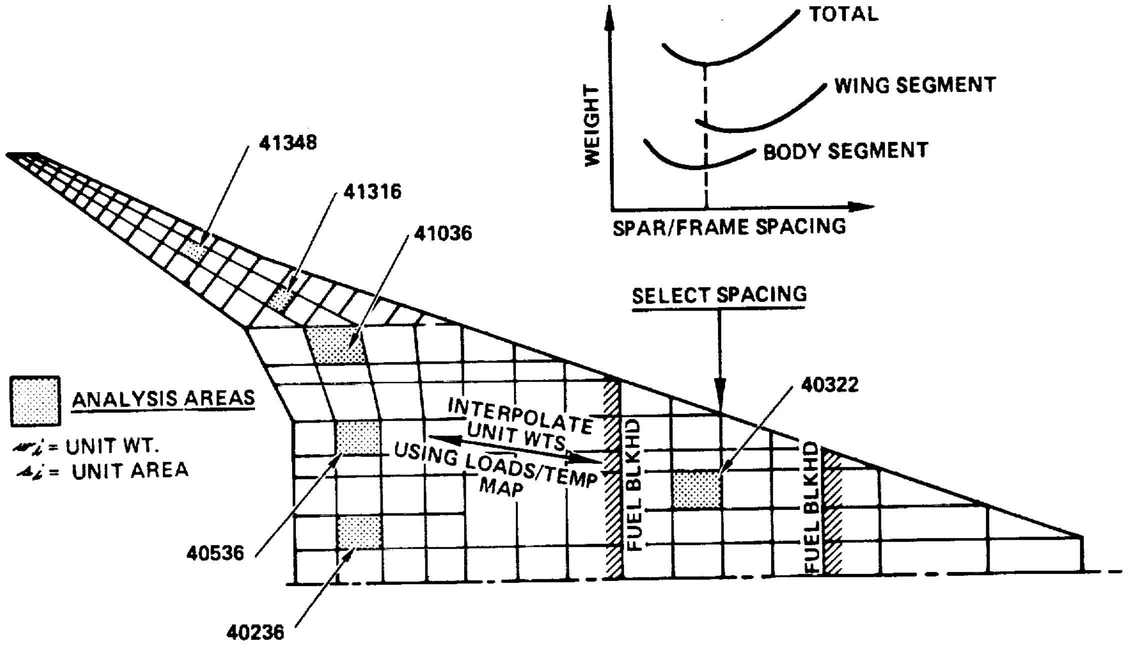
In the locations where wing spars mate to the body frames, the spacing is selected by the minimum weight combination of the wing and body segments.

Wing. - To obtain the basic wing box structure weight, the unit weights are integrated over the entire wing box areas as pictorially displayed in Figure 15-9. Assessment is also made of special structural items, access doors, systems provisions and other non-optimum items. The summation of the basic box structure, control surfaces, leading and trailing edges, result in the total wing group weight.

Fuselage. - In a similar manner to that described for the wing, four body analysis areas were selected to investigate the three candidate structural concepts of Task I. Interpolation between these areas provide sufficient information to derive a total basic shell structural weight. Special structural features are added to the basic shell structure to derive the total body weight. The procedure employed is outlined in Figure 15-10.

WING STRUCTURE MASS-INITIAL SCREENING

Analysis regions for the initial screening of the candidate structural concepts are indicated in Figure 15-9. They are described as follows:


TOTAL
WING SEGMENT
BODY SEGMENT
WEIGHT
SPAR/FRAME SPACING
SELECT SPACING
41348
41316
41036
ANALYSIS AREAS
w_i = UNIT WT.
s_i = UNIT AREA
INTERPOLATE UNIT WTS USING LOADS/TEMP MAP
FUEL BLKHD
FUEL BLKHD
40322
40536
40236

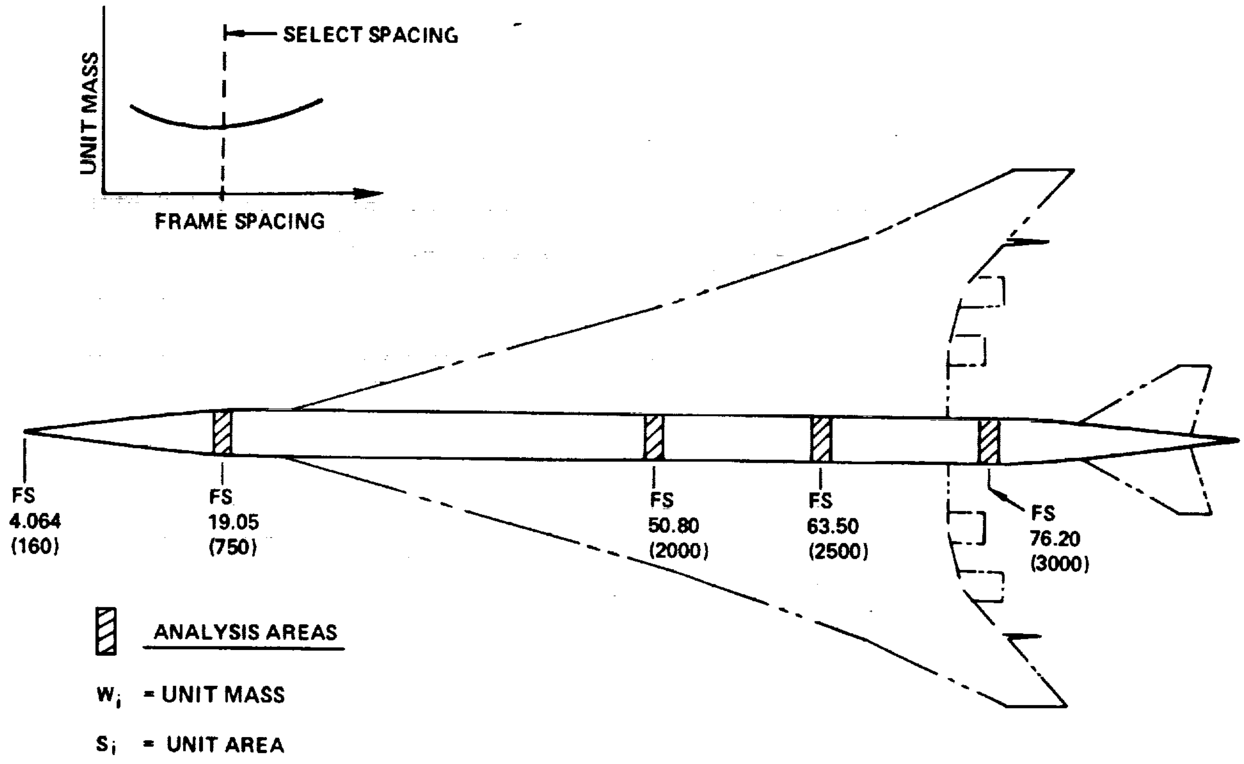
$\sum_{i}^{n} w_i\, s_i$ = BASIC BOX STRUCTURE MASS

PLUS: BODY & OUTB'D JOINT RIBS, MLG WELL STR, ENG. & V. FIN RIB
FLAP, AILERON, SPOILER SUPT. STR.
FUEL BULKHEADS, ACCESS DOORS
JACK FTG'S, FAIRINGS, FILLETS, SEAL. & FINISH
LOCAL REINF., FASTENERS, SHIMS & CLIPS
PROVISIONS FOR SYSTEMS

TOTAL = BOX STRUCTURE MASS
PLUS: L.E. & FLAPS
T.E. & FLAPS, SPOILERS, AILERONS
BODY FAIRING
MLG DOORS

TOTAL = WING MASS

Figure 15-9. Wing Mass Estimation Methodology


SELECT SPACING
UNIT MASS
FRAME SPACING
FS 4.064 (160)
FS 19.05 (750)
FS 50.80 (2000)
FS 63.50 (2500)
FS 76.20 (3000)
ANALYSIS AREAS
W_i = UNIT MASS
S_i = UNIT AREA

$\sum_{i}^{n} W_i S_i$ = BASIC SHELL STRUCTURE MASS

PLUS:
- NOSE AND FLIGHT STATION
- NOSE LANDING GEAR WELL
- WINDSHIELD AND WINDOWS
- FLOORING AND SUPPORTS
- DOORS AND MECHANISM
- UNDER WING FAIRING
- CARGO PROVISIONS
- WING/BODY FRAMES AND FITTINGS
- TAIL/BODY FRAMES AND FITTINGS
- PROV. FOR SYSTEMS
- FINISH AND SEALANT

TOTAL = FUSELAGE MASS

Figure 15-10. Fuselage Mass Estimation Methodology

Point Design Region	Location	Area (Sq. Ft.)
40322	Forward Box	44.4
40536	Aft Box	35.6
41348	Tip Box	10.3

By interpolation from the analysis point design regions, unit weights for each concept are applied to the panel areas shown in Figure 15-11 to derive the total box structural weight.

The initial screening data includes a non-optimum allowance for surface-to-cap joints of approximately 4-percent. Additional non-optimum allowances are applied to the box weight to arrive at a typical estimate of the "as-constructed" weight. These non-optimum allowances are itemized as:

	Non optimum Factor (NOF)
Joints and splices to surface panels	7-percent
Margins of safety (average)	3-percent
Sheet tolerances	2-percent
System provisions (Electrical, Fuel Controls)	5-percent
Access provisions (one surface only)	6-percent
Finish, sealant, misc.	3-percent
Total NOF	26-percent

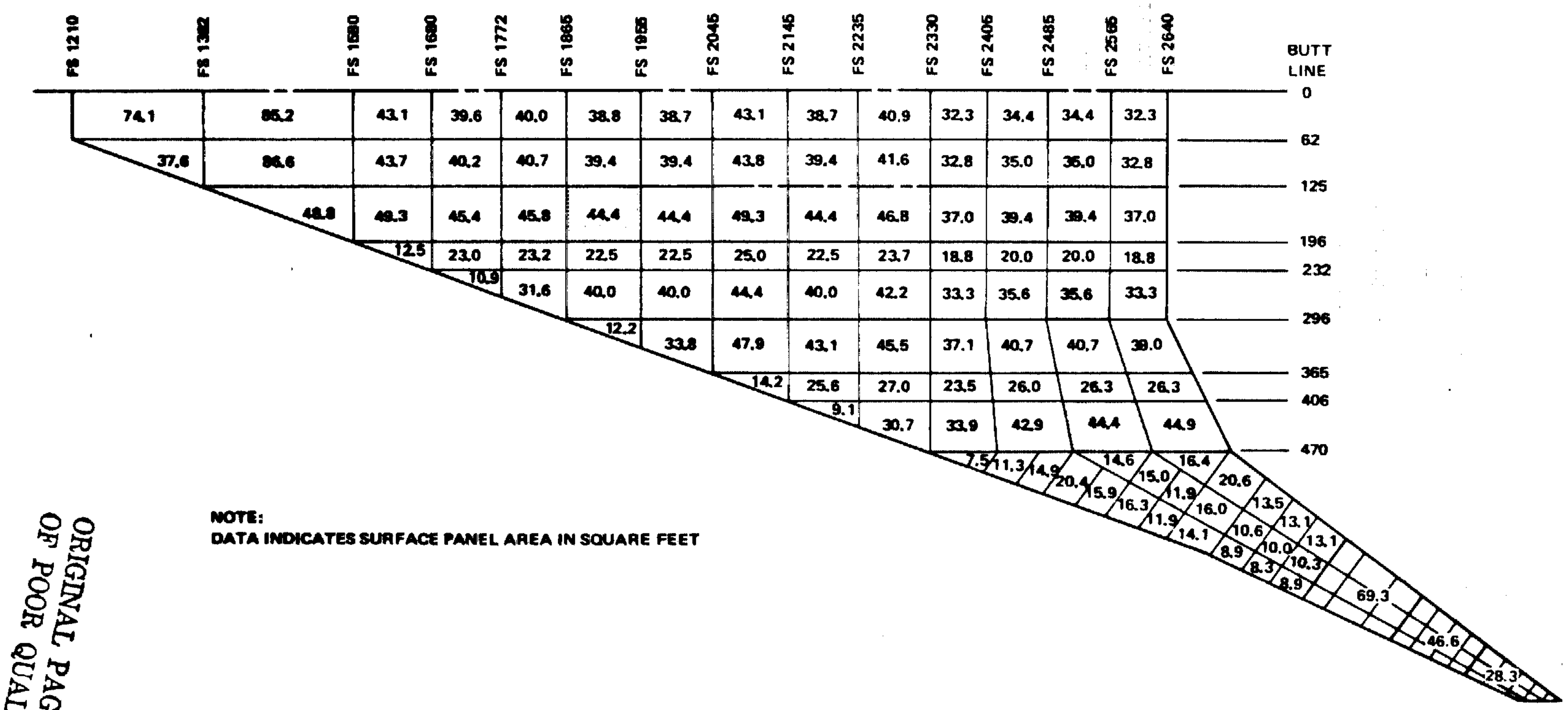
Use of these allowances, for example, means that a stress analysis which indicates a five pound-per-square-foot panel yields an estimated fabricated weight of (1.26 x 5.0) = 6.3 pounds per square foot.

For comparison purposes, the wing weight was divided into two major categories:

- Variable weight
- Fixed weight

The variable weight consists of that portion of the box structure which is influenced by the structural concept being considered, such as the upper and lower surfaces and intermediate ribs and spars.

The fixed weight consists of those items which are unaffected by box structural concept, such as main landing gear provisions, surface controls, engine support structure, leading and trailing edge structure.


FS 1210
FS 1382
FS 1580
FS 1680
FS 1772
FS 1865
FS 1955
FS 2045
FS 2145
FS 2235
FS 2330
FS 2406
FS 2485
FS 2565
FS 2640
BUTT LINE
0
62
125
196
232
296
365
406
470
74.1
85.2
43.1
39.6
40.0
38.8
38.7
43.1
38.7
40.9
32.3
34.4
34.4
32.3
37.6
86.6
43.7
40.2
40.7
39.4
39.4
43.8
39.4
41.6
32.8
35.0
36.0
32.8
48.8
48.3
45.4
45.8
44.4
44.4
49.3
44.4
46.8
37.0
39.4
39.4
37.0
12.5
23.0
23.2
22.5
22.5
25.0
22.5
23.7
18.8
20.0
20.0
18.8
10.9
31.6
40.0
40.0
44.4
40.0
42.2
33.3
35.6
35.6
33.3
12.2
33.8
47.9
43.1
45.5
37.1
40.7
40.7
39.0
14.2
25.6
27.0
23.5
26.0
26.3
26.3
9.1
30.7
33.9
42.9
44.4
44.9
7.5
11.3
14.9
20.4
14.6
15.9
16.3
15.0
11.9
11.9
16.4
16.0
14.1
20.6
13.5
10.6
8.9
13.1
10.0
8.3
13.1
10.3
8.9
69.3
46.6
28.3

NOTE:
DATA INDICATES SURFACE PANEL AREA IN SQUARE FEET

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Figure 15-11. Wing Box Areas - Task I

The least weight concept for each stiffening arrangement is listed in Table 15-21. From this table, it appears that the convex beaded, chordwise stiffened arrangement with composite reinforced spar cap is preferred throughout the wing box. Subsequent flutter optimization resulted in the monocoque arrangement to be preferred for the tip structure from the least weight viewpoint.

Chordwise Stiffened Design Concepts

The chordwise stiffened designs employ surface panel concepts that have stiffening elements oriented in the chordwise direction. The substructure arrangement is essentially a multispar structure with widely spaced ribs. Submerged caps are provided except at panel closeouts and at fuel tank bulkheads. Four surface panel concepts were considered (see Section 1 Structural Design Concepts):

- Circular-arc concave beaded skin
- Circular-arc convex beaded skin
- Trapezoidal corrugation-concave beaded skin
- Beaded corrugation - concave beaded skin

The resulting wing weights are summarized in Table 15-22. The convex beaded concept was found to be significantly lighter than the others evaluated. In all cases, the spar weights are relatively heavy since the surface panels are ineffective in transmitting the wing span bending loads.

A general expression for deriving box panel unit weights, using the three analysis point design regions as a starting point, is based on the following parameters:

- Inplane loads: N_x, N_y, N_{xy}
- Pressure loads: Δp

The expression for the convex beaded optimum panel weight is:

$$w(\text{lb/ft}^2) = \left[\left(|N_x| + |N_y| + |2\,N_{xy}| + 350|\Delta p|\right) \div (2190)\right]$$

Where,

N_x = axial chordwise load (lb/in)

N_y = axial spanwise load (lb/in)

N_{xy} = panel shear flow (lb/in)

TABLE 15-21. SUMMARY OF WING MASS - INITIAL SCREENING

PLAN AREA (ft^2)	ITEM	CHORDWISE CONVEX BEADED	SPANWISE HAT STIFFENED	MONOCOQUE H/C SAND. WELDED	CHORDWISE COMPOSITE REINFORCED
	VARIABLE WEIGHT (A)	56,655	62,176	50,796	43,624
1231	CENTER SECTION (BL 0→62)	(8,722)	(9,380)	(8,274)	(6,716)
	UPPER SURFACE	1,570	3,377	2,482	1,570
	LOWER SURFACE	1,570	3,518	2,532	1,570
	SPARS	4,884	1,041	2,325	2,878
	RIBS	693	1,444	935	698
5038	INTERM. PANEL (BL 62→470)	(39,296)	(43,478)	(35,514)	(30,258)
	UPPER	7,073	15,652	10,654	7,073
	LOWER	7,073	16,304	10,867	7,073
	SPARS	22,006	4,826	9,980	12,963
	RIBS	3,144	6,696	4,013	3,144
896	OUTER PANEL (BL 470→TIP)	(8,637)	(9,318)	(7,008)	(6,650)
	UPPER	1,555	3,355	2,102	1,555
	LOWER	1,555	3,494	2,145	1,555
	SPARS	4,837	1,034	1,969	2,850
	RIBS	690	1,435	792	690
	FIXED WEIGHT	(41,352)	←	→	(41,352)
1047	LEADING EDGE	5,235			5,235
1955	TRAILING EDGE	4,888			4,888
	BL 62 RIBS	1,430			1,430
	BL 470 RIBS	700			700
	FIN ATTACH RIBS	435			435
	REAR SPAR	3,400			3,400
	ENG. SUP'T STRUCTURE	3,580			3,580
484	MLG DOORS	2,904			2,904
	– WHEEL WELL AND ATTACH.	3,750			3,750
800	WING/BODY FAIRING	1,600			1,600
133	LE FLAPS	1,130			1,130
553	TE FLAPS	5,890			5,890
250	AILERONS	1,250			1,250
225	SPOILERS	1,360			1,360
	FUEL BULKHEADS	3,800			3,800
	TOTAL (STRENGTH DESIGN ONLY)	98,007	103,528	92,148	84,976

(A) BASED ON 20-INCH SPAR OR RIB SPACING

TABLE 15-22. ESTIMATED WING MASS - CHORDWISE STIFFENED CONCEPT - INITIAL SCREENING

ITEM	CONCAVE BEADED	CONVEX BEADED	TRAPEZOID NO BEAD	TRAPEZOID BEADED
VARIABLE WEIGHT:	(58,660)	(56,655)	(60,236)	(61,743)
CENTER SECTION	(9,030)	(8,722)	(9,273)	(9,505)
UPPER SURFACE	1,716	1,570	1,762	1,901
LOWER SURFACE	1,716	1,570	1,762	1,901
SPAR CAPS AND WEBS	4,876	4,884	5,007	5,038
RIBS	722	698	742	665
INTERM. PANEL (BL. 62 → 470)	(40,687)	(39,296)	(41,780)	(42,825)
UPPER SURFACE	7,730	7,073	7,938	8,565
LOWER SURFACE	7,730	7,073	7,938	8,565
SPARS	21,970	22,006	22,561	22,697
RIBS	3,257	3,144	3,343	2,998
OUTER PANEL (BL. 470 → TIP)	(8,943)	(8,637)	(9,183)	(9,413)
UPPER SURFACE	1,699	1,555	1,745	1,883
LOWER SURFACE	1,699	1,555	1,745	1,883
SPARS	4,829	4,837	4,959	4,989
RIBS	716	690	734	658
FIXED WEIGHT:	(41,352)	←	→	(41,352)
LEADING EDGE	5,235			5,235
TRAILING EDGE	4,888			4,888
B.L. 62 RIBS	1,430			1,430
B.L. 470 RIBS	700			700
FIN ATTACH RIBS	435			435
REAR SPAR	3,400			3,400
ENG. SUP'T. STRUCTURE	3,580			3,580
MLG DOORS	2,904			2,904
WHEEL WELL AND ATTACH.	3,750			3,750
WING/BODY FAIRING	1,600			1,600
LE FLAPS	1,130			1,130
TE FLAPS	5,890			5,890
AILERONS	1,250			1,250
SPOILERS	1,360			1,360
FUEL BULKHEADS	3,800			3,800
TOTAL (STRENGTH DESIGN ONLY)	100,012	98,007	101,588	103,095

$\Delta p = \left(\left|\Delta p_u\right| + \left|\Delta p_\ell\right|\right)$, pressure load for combined upper and lower surfaces (lb/in^2)

The total variable weight, then, after interpolating to determine each box panel unit weight is:

$$W_v = \left(\sum_1^n w_i S_i\right)(1.26) \qquad \text{(Reference Figure 15-9)}$$

For the convex beaded concept:

$$W_v = 56{,}655 \text{ pounds}; \; w_{ave} = \frac{W_v}{S_{box}} = \frac{56{,}655 \text{ (lb)}}{7{,}165 \text{ (ft}^2\text{)}} = 7.91 \text{ psf}$$

This result is based on the unit weights for the three analysis regions as tabulated below:

PANEL CONCEPT	Point Design Regions(A)			w_{ave} (incl. 1.26 NOF)
	40322	40536	41348	
	Unit Weights ~ Pound per square foot			
Concave-Beaded	4.10	11.45	9.85	8.19
Convex-Beaded	3.80	11.30	9.75	7.91
Trapezoidal Corrugation	4.35	11.55	9.90	8.41
Beaded Corrugation	4.60	11.60	10.00	8.62

(A) Unit Weights do not include non optimum factor (NOF)

The total variable weight for the other concepts was facilitated by deriving a general expression, where:

$$W_v = \left(\frac{S_{box}}{5.17}\right)\left[3\left(w_{40322}\right) + \left(w_{40536}\right) + \left(w_{41348}\right)\right] 1.26$$

Temperature variation between panels is small and has negligible effect on the process of weight interpolation, since the structural concept for each point design already accounts for the effects of thermal stresses. For this reason, weight interpolation has been performed as a function of inplane loads and normal pressure loads.

Spanwise Stiffened Design Concepts

The spanwise stiffened wing concept is a multirib design with closely spaced ribs and widely spaced spars. The surface panel configurations have effective load carrying capability in their stiffened (span) direction. Smooth skins are required for aerodynamic performance. The four spanwise stiffened designs investigated are as follows:

- Zee stiffened
- Integral zee stiffened
- Hat stiffened
- Integral stiffened

Their comparative weights are summarized in Table 15-23. As indicated on the table, the hat stiffener concept is the least weight. In all cases, the spar weights are relatively light, compared to the chordwise stiffened designs due to the ability of the surfaces to carry spanwise inplane loads.

A general expression for deriving box panel unit weight is:

$$w(\mathrm{lb/ft}^2) = \left[\left(|N_x| + |N_y| + |2N_{xy}| + 350|\Delta p|\right) \div (2190)\right]$$

where:

N_x = axial chordwise load (lb/in)

N_y = axial spanwise load (lb/in)

N_{xy} = panel shear flow (lb/in)

Δp = pressure load ($\mathrm{lb/in}^2$)

Unit weights for each analysis panel of the four structural concepts are from Section 12 and are summarized below:

Panel Concept	Point Design Region$^{(A)}$			w_{ave} (incl. 1.26 NOF)
	40322	40536	41348	
	Unit Weight ~ Pound Per Square Foot (psf)			
Zee stiffened	4.95	13.30	8.55	8.77
Integral zee	4.70	13.80	8.50	8.69
Hat stiffened	4.70	13.75	8.50	8.68
Integral stiffened	5.40	14.25	9.65	9.58

(A) Unit weights do not include non optimum factor (NOF)

TABLE 15-23. ESTIMATED WING MASS - SPANWISE STIFFENED CONCEPT - INITIAL SCREENING

ITEM	ZEE STIFFENED	INTEGRAL ZEE	HAT STIFFENED	INTEGRAL STIFFENED
VARIABLE WEIGHT	(62,827)	(62,248)	(62,176)	(68,601)
CENTER SECTION	(9,479)	(9,393)	(9,380)	(10,353)
UPPER SURFACE	3,659	3,428	3,377	4,286
LOWER SURFACE	3,327	3,475	3,518	3,572
SPAR CAPS AND WEBS	1,043	1,043	1,041	1,046
RIBS	1,450	1,447	1,444	1,449
INTERM. PANEL (BL 62 → 470)	(43,931)	(43,528)	(43,478)	(47,962)
UPPER SURFACE	16,957	15,888	15,652	19,856
LOWER SURFACE	15,420	16,105	16,304	16,547
SPARS	4,832	4,832	4,826	4,844
RIBS	6,722	6,703	6,696	6,715
OUTER PANEL (BL 470 → TIP)	(9,417)	(9,327)	(9,318)	(10,286)
UPPER SURFACE	3,635	3,404	3,355	4,258
LOWER SURFACE	3,305	3,451	3,494	3,549
SPARS	1,036	1,036	1,034	1,039
RIBS	1,441	1,436	1,435	1,440
FIXED WEIGHT	(41,352)	←	→	(41,352)
LEADING EDGE	5,235			5,235
TRAILING EDGE	4,883			4,883
B.L. 62 RIBS	1,430			1,430
B.L. 470 RIBS	700			700
FIN ATTACH RIBS	435			435
REAR SPAR	3,400			3,400
ENG. SUP'T. STRUCTURE	3,580			3,580
MLG DOORS	2,904			2,904
MLG WHEEL WELL AND ATTACH.	3,750			3,750
WING/BODY FAIRING	1,600			1,600
LE FLAPS	1,130			1,130
TE FLAPS	5,890			5,890
AILERONS	1,250			1,250
SPOILERS	1,360			1,360
FUEL BULKHEADS	3,800			3,800
TOTAL (STRENGTH DESIGN ONLY)	104,179	103,600	103,528	109,953

Total variable weight after interpolating to determine each box panel weight is:

$$W_v = \left(\sum_{1}^{n} w_i \; s_i \right) (1.26)$$

For the hat stiffener concept, total variable weight is 62,176 pounds. This results in an average box unit weight of 8.68 pounds per square foot:

$$w_{ave}(lb/ft^2) = \left[(62{,}176) \div (7165) \right] = 8.68$$

Total variable weight for the other structural concepts was facilitated by deriving a general expression where:

$$W_v = \left(\frac{S_{box}}{5.275} \right) \left[\left(3 \; w_{40322} + w_{40536} + w_{41348} \right) 1.26 \right]$$

The weight distribution between center, intermediate and tip box structure was taken to be proportional to that found in the hat stiffened concept. Weight distribution between surfaces, ribs and spar is based on the structural analysis data of Section 12.

Monocoque Design Concepts

The monocoque construction consists of biaxially stiffened panels which support the principal load in both the span and chord direction. For the substructure arrangement, both multirib and multispar designs were evaluated. The initial screening and detailed analysis mass estimation of these concepts were performed concurrently and are reported in Monocoque Wing Design section.

WING STRUCTURE MASS-DETAILED CONCEPT ANALYSIS

As a result of the initial screening process, each of the most promising concepts were investigated further through the analysis of three additional point design regions. The results of this analysis effort is summarized in Table 15-24.

TABLE 15-24. SUMMARY OF WING MASS - DETAILED CONCEPT ANALYSIS

ARRANGEMENT	CHORDWISE			SPANWISE	MONOCOQUE		HYBRID
SURFACE PANEL	CONVEX-BEADED		HAT	HAT	HONEYCOMB SAND.		BEST COMB-NATION
MATERIAL SYSTEM	TI-6Al-4V	COMPOSITE REINF.		TI-6Al-4V	TI-6Al-4V		
CONCEPT NO. (ASSET)	①	⑤	⑥	②	③	④	⑦
VARIABLE WEIGHT	64,658	48,082	53,487	63,482	50,978	53,794	47,268
FORWARD BOX	22,090	20,580	24,184	25,364	21,982	24,057	20,580
SURFACES	9,545	9,452	14,655	15,842	14,656	14,386	
SPARS	9,975	8,558	6,959	3,913	4,616	5,965	
RIBS	2,570	2,570	2,570	5,609	2,710	3,706	
AFT BOX (FS 2330 TO 2640)	29,016	17,384	18,592	25,242	19,692	20,153	17,384
SURFACES	7,622	7,302	9,225	20,947	13,984	13,824	
SPARS	19,880	8,568	7,853	2,243	4,060	4,416	
RIBS	1,514	1,514	1,514	2,052	1,648	1,913	
TIP BOX (BL 470 TO TIP)	13,552	10,118	10,711	12,876	9,304	9,584	9,304
SURFACES	6,464	6,397	7,166	10,965	8,059	8,059	
SPARS	6,405	3,038	2,862	914	928	1,044	
RIBS	683	683	683	997	317	481	
TOTAL REINF. COMPOSITE		(5,480)	(10,668)				
FIXED WEIGHT	(41,352)						(41,352)
LEADING EDGE	5,235						5,235
TRAILING EDGE	4,888						4,888
B.L. 62 RIBS	1,430						1,430
B.L. 470 RIBS	700						700
FIN ATTACH RIBS	435						435
REAR SPAR	3,400						3,400
ENG. SUP'T. STRUCT.	3,580						3,580
MLG ~ DOORS	2,904						2,904
~SUP'T. STRUCT	3,750						3,750
WING/BODY FAIRING	1,600						1,600
LE FLAPS	1,130						1,130
TE FLAPS	5,890						5,890
AILERONS	1,250						1,250
SPOILERS	1,360						1,360
FUEL BULKHEADS	3,800						3,800
TOTAL WING WEIGHT	106,010	89,434	94,839	104,834	92,330	95,146	88,620

NOTES:

1. ASSEMBLY JOINING FOR ALL CONCEPTS (EXCEPT CONCEPT ④) IS MECHANICALLY FASTENED. CONCEPT ④ IS WELDED.
2. CONCEPT ⑤ – COMPOSITE REINFORCED (B/PI) SPAR CAPS ONLY
3. CONCEPT ⑥ – COMPOSITE REINFORCED SPAR CAPS AND SURFACE PANELS

For all-metallic construction, the mechanically fastened monocoque concept is least weight. However, the application of composite reinforcing (boron-polyimide) to the spar caps of the chordwise stiffened concept makes its variable weight lowest by almost 3000 pounds per aircraft. Furthermore, the best combination from a weight standpoint is an all-metallic, mechanically fastened-monocoque design for the wing tip structure with the forward and aft boxes constructed of convex beaded, chordwise stiffened surface panels with composite reinforced spar caps.

Chordwise Stiffened Wing Design

Analysis results of the 3 additional point design regions plus the three point design regions used in the initial screening are used to provide a better basis for evaluation of the variable weights of the wing box weight. The additional regions are described as follows:

Point Design Region	Location	Area (Ft^2)
40236	Aft box	35.0
41036	Aft box	44.4
41316	Tip	11.9

The loads and unit weights for all six point design regions are compared in Table 15-25. The optimum unit weight from stress analysis is compared to the estimated unit weight derived from the modified loading parameter equation. This equation was used to calculate unit weights for the remaining box panels and re-evaluate the total variable weight.

Panel weights resulting from the detailed concept analysis are shown in Figure 15-12. These are optimum weights based on strength requirements only.

Fail-safe requirements for each point region are shown in Table 15-26. This data was converted to an average fail-safe penalty for each of the three point designs. Figure 15-13 indicates that the fail-safe increment is primarily applied to the spar web and clips (85-percent) with the remainder (15-percent) applied to the surface panels.

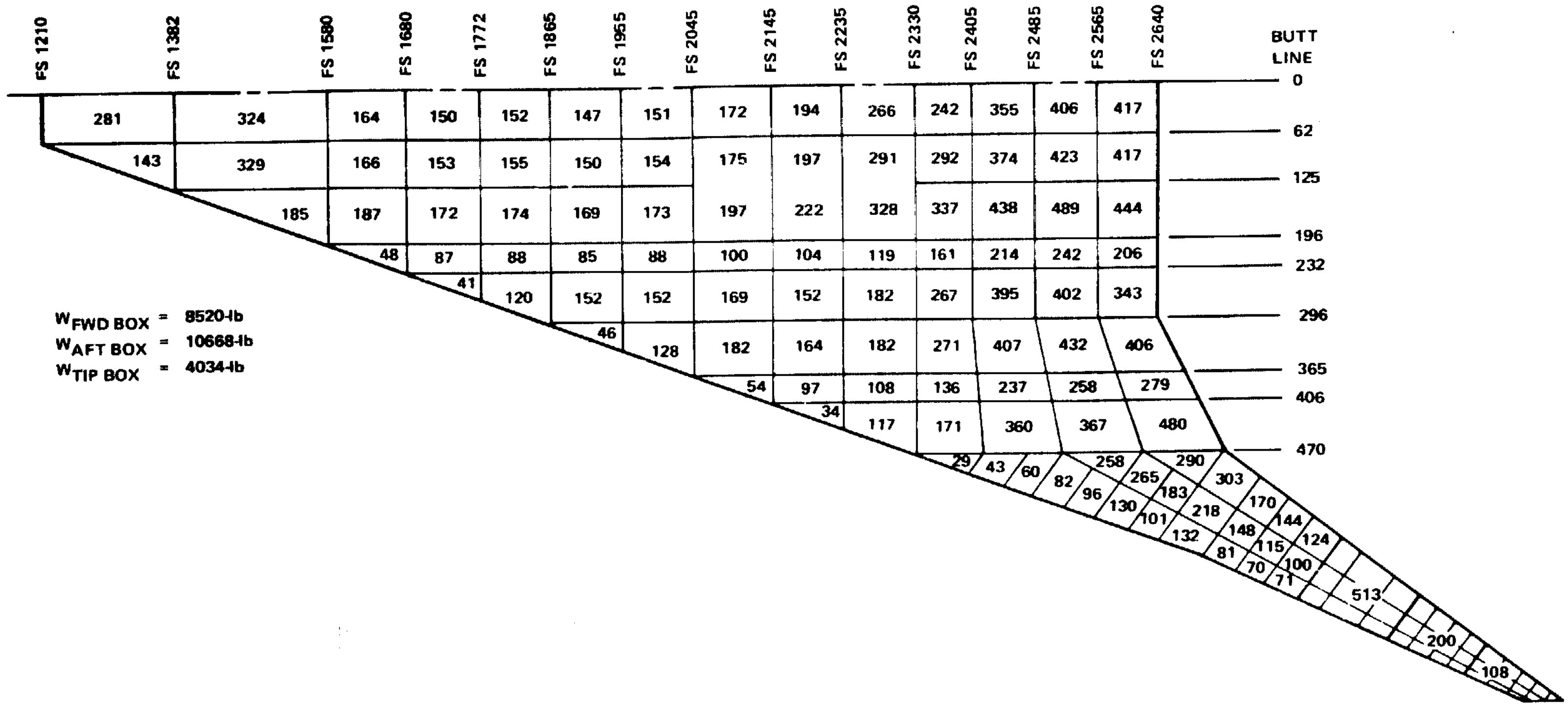
Flutter suppression requires the addition of the increment shown in Figure 15-14. For the chordwise stiffened design 2938 pounds per aircraft is required.

TABLE 15-25. PANEL LOAD AND UNIT WEIGHT - CHORDWISE STIFFENED DESIGN

POINT DESIGN REGION	INPLANE LOADS N_X N_Y N_{XY} (lb/in.)	PRESSURE LOADS UPR. LWR. $\sum$ (lb/in^2)	PANEL UNIT WEIGHTS TOTAL UPR LWR SPARS RIBS (lb/ft^2)	PANEL ESTIMATED WEIGHTS W_{PANEL} (A) (lb/ft^2)
40322	488 −1,063 −120	−8.33 −9.47 17.80	3.80 0.83 0.94 1.53 (B) 0.50	3.82
40236	658 −16,387 −1,316	−8.96 −10.14 19.10	12.08 1.03 1.25 9.15 (B) 0.65	12.55
40536	−1,305 −14,379 −2,354	−7.47 −8.29 15.76	11.30 1.61 1.34 7.75 (B) 0.60	12.33
41036	−1,442 −9,156 −2,237	−1.27 0.11 1.38	8.25 1.35 1.45 4.95 0.50 (B)	7.40
41316	571 −16,982 +4,807	4.98 −0.26 5.24	15.38 2.60 2.05 10.13 0.60 (B)	13.81
41348	−1,433 −10.800 2,483	−5.07 1.0 6.07	9.75 1.63 1.32 6.20 0.60 (B)	9.20

(A) $W_{PANEL} = \left[|N_X| + |N_Y| + |2N_{XY}| + 350|\Delta \rho| \right] \div 2100$

(B) INCLUDES STRESS NON OPTIMUM FACTOR


FS 1210
FS 1382
FS 1580
FS 1680
FS 1772
FS 1865
FS 1965
FS 2045
FS 2145
FS 2235
FS 2330
FS 2405
FS 2485
FS 2565
FS 2640
BUTT LINE
0
62
125
196
232
296
365
406
470
281
324
164
150
152
147
151
172
194
266
242
355
406
417
143
329
166
153
155
150
154
175
197
291
292
374
423
417
185
187
172
174
169
173
197
222
328
337
438
489
444
48
87
88
85
88
100
104
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161
214
242
206
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182
267
395
402
343
46
128
182
164
182
271
407
432
406
54
97
108
136
237
258
279
34
117
171
360
367
480
29
43
60
82
96
130
101
132
258
265
183
218
290
303
170
148
81
144
115
70
124
100
71
513
200
108
W_FWD BOX = 8520-lb
W_AFT BOX = 10668-lb
W_TIP BOX = 4034-lb

Figure 15-12. Optimum Panel Weights (lb/side) - Chordwise Stiffened Design

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TABLE 15-26. COMPONENT WEIGHT DERIVATION - CHORDWISE STIFFENED DESIGN - DETAILED CONCEPT ANALYSIS

PANEL NO.	UNIT WEIGHTS (psf)				
	SURFACES	SPARS	FAIL-SAFE (B)	RIBS	TOTAL
40322	1.77	1.53	(0.10)	0.50	3.80
40236	2.28	9.15	(1.75)	0.65	12.08
40536	2.95	7.75	(0.93)	0.60	11.30
41036	2.80	4.95	(0.56)	0.50	8.25
41316	4.65	10.13	(1.00)	0.60	15.38
41348	2.95	6.20	(0.41)	0.60	9.75

			UNIT WT. (psf)	BOX WT. (lb)
FWD. BOX (AREA = 4136.6 ft^2/AIRCRAFT)		TOTAL =	5.340	22,090
		FAIL-SAFE		
SURFACES	0.122 (1.77 X 9 + 2.80) +	0.0225	2.3075	9,545
SPARS	0.122 (1.53 X 9 + 4.95) +	0.1275	2.4113	9,975
RIBS	0.122 (0.5 X 9 + 0.5) +	0.0112	0.6212	2,570
AFT BOX (AREA = 2132.4 ft^2/AIRCRAFT)		TOTAL =	13.067	29,016
		FAIL-SAFE		
SURFACES	0.3162 (2.28 + 2.95 + 2 X 2.80) +	0.150 =	3.574	7,622
SPARS	0.3162 (9.15 + 7.75 + 2 X 4.95) +	0.850 =	9.323	19,880
RIBS	0.3162 (0.65 + 0.60 + 2 X 0.50) +	0	0.710	1,514
TIP BOX (AREA = 896 ft^2/AIRCRAFT)		TOTAL =	15.125	13,552
		FAIL-SAFE		
SURFACES	0.3465 (4.65 + 2.95 + 2 X 1.77) +	0.075 =	3.935	3,526
	FLUTTER INCREMENT	=	3.279	+2,938
SPARS	0.3465 (10.13 + 6.20 + 2 X 1.53) +	0.430 =	7.149	6,405
RIBS	0.3465 (0.6 + 0.6 + 2 X 0.5) +		0.762	683

(A) 20-INCH SPAR SPACING

(B) WEIGHT INCLUDED IN SURFACES, SPARS, AND RIBS

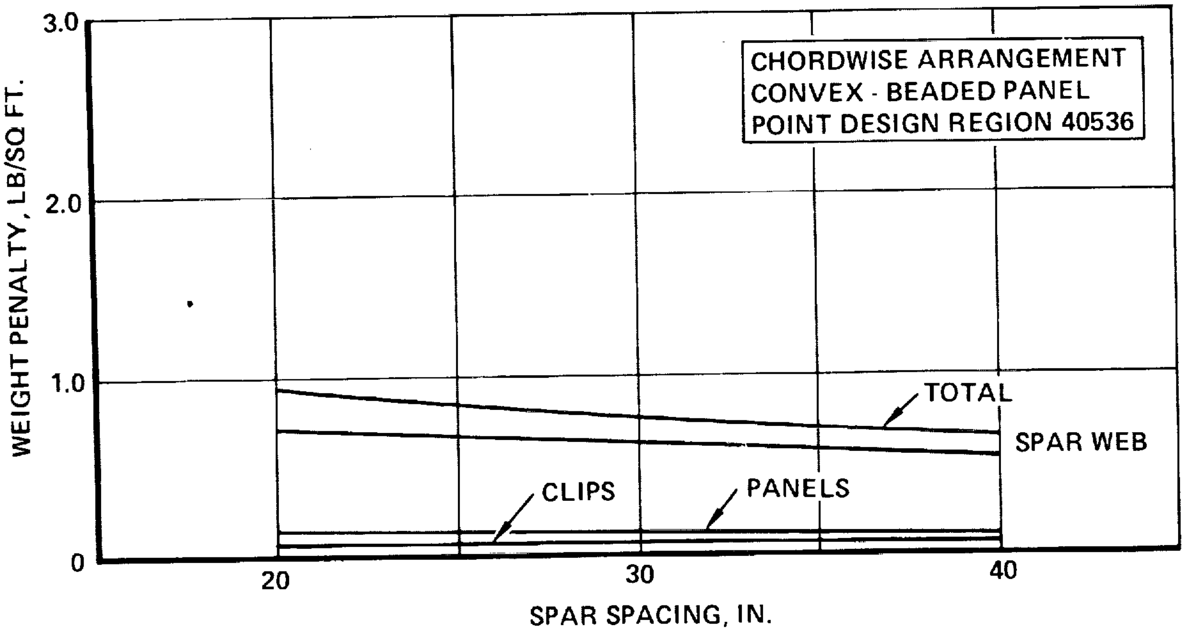


Figure 15-13. Component Weight Penalties for a Damaged Spar Cap Chordwise Arrangement

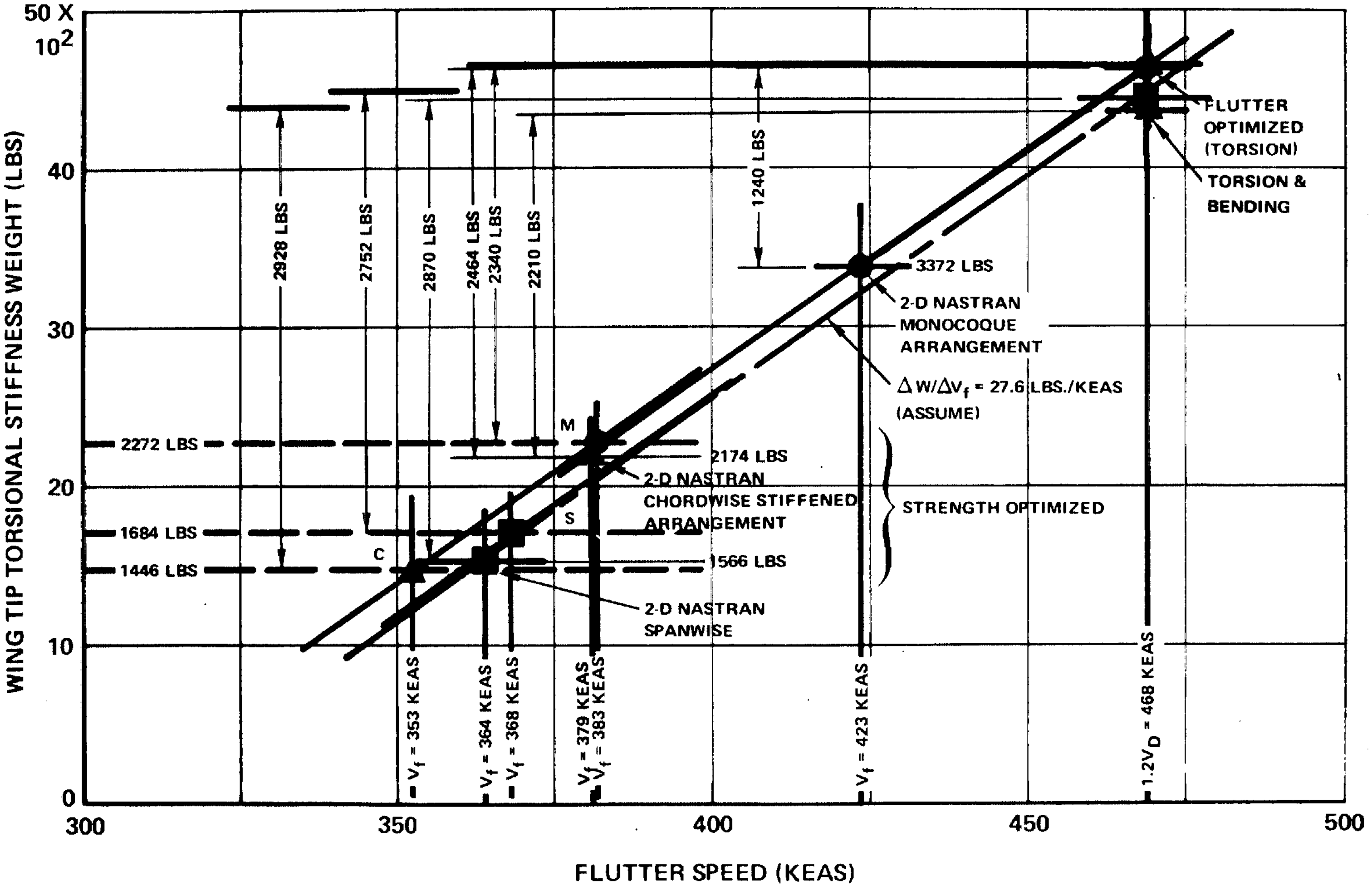


Figure 15-14. Flutter Weight Increment Required Over Strength Design

A build-up to total box weight, including the above increments is tabulated in Figure 15-15. Cumulative plots of the "strength" weight versus span are also shown.

From the total box weight of Figure 15-15, equations were developed to express the average box unit weight in terms of the detailed stress analysis point design region unit weight:

$$w_{\text{fwd box}} \left(\text{lb/ft}^2\right) = 0.1220 \left(9w_{40322} + w_{41036}\right)$$

$$w_{\text{aft box}} \left(\text{lb/ft}^2\right) = 0.3162 \left(w_{40236} + w_{40536} + 2w_{41036}\right)$$

$$w_{\text{tip box}} \left(\text{lb/ft}^2\right) = 0.3465 \left(w_{41316} + w_{41348} + 2w_{40322}\right)$$

These equations are used in Table 15-26 to derive the component weight breakdown shown in Table 15-24.

Spanwise Stiffened Wing Design

The detailed concept analysis weights for the hat stiffened concept are obtained by incorporating into the estimation procedure, the results of the 3 additional point design regions (40236, 41036 and 41316).

The loads and unit weights for all 6 point design regions are compared in Table 15-27. The optimum unit weight from stress analysis are compared to the estimated unit weight derived from the modified loading parameter equation. This equation (Table 15-27) was used to calculate the unit weights for the remaining box panels and to reevaluate the total variable weight.

Panel weights resulting from this analysis are shown in Figure 15-16. These are optimum weights based on strength requirements only. No fail-safe increments are required for the spanwise stiffened design.

Flutter suppression requires the addition of the increment shown in Figure 15-14 to the wing tip structure. For the spanwise stiffened design 2928 pounds per aircraft is required to achieve the required flutter margin.

CHORDWISE STIFFENED - CONVEX BEADED

	WEIGHT/SIDE (LB.)		
	FWD. BOX	AFT BOX	TIP BOX
STRENGTH ONLY	8,520	10,668	4,034
NON-OPTIMUM	+2,215	+2,774	+1,049
FAIL-SAFE	+310	+1,066	+224
FLUTTER	–	–	+1,469
TOTAL	11,045	14,508	6,776

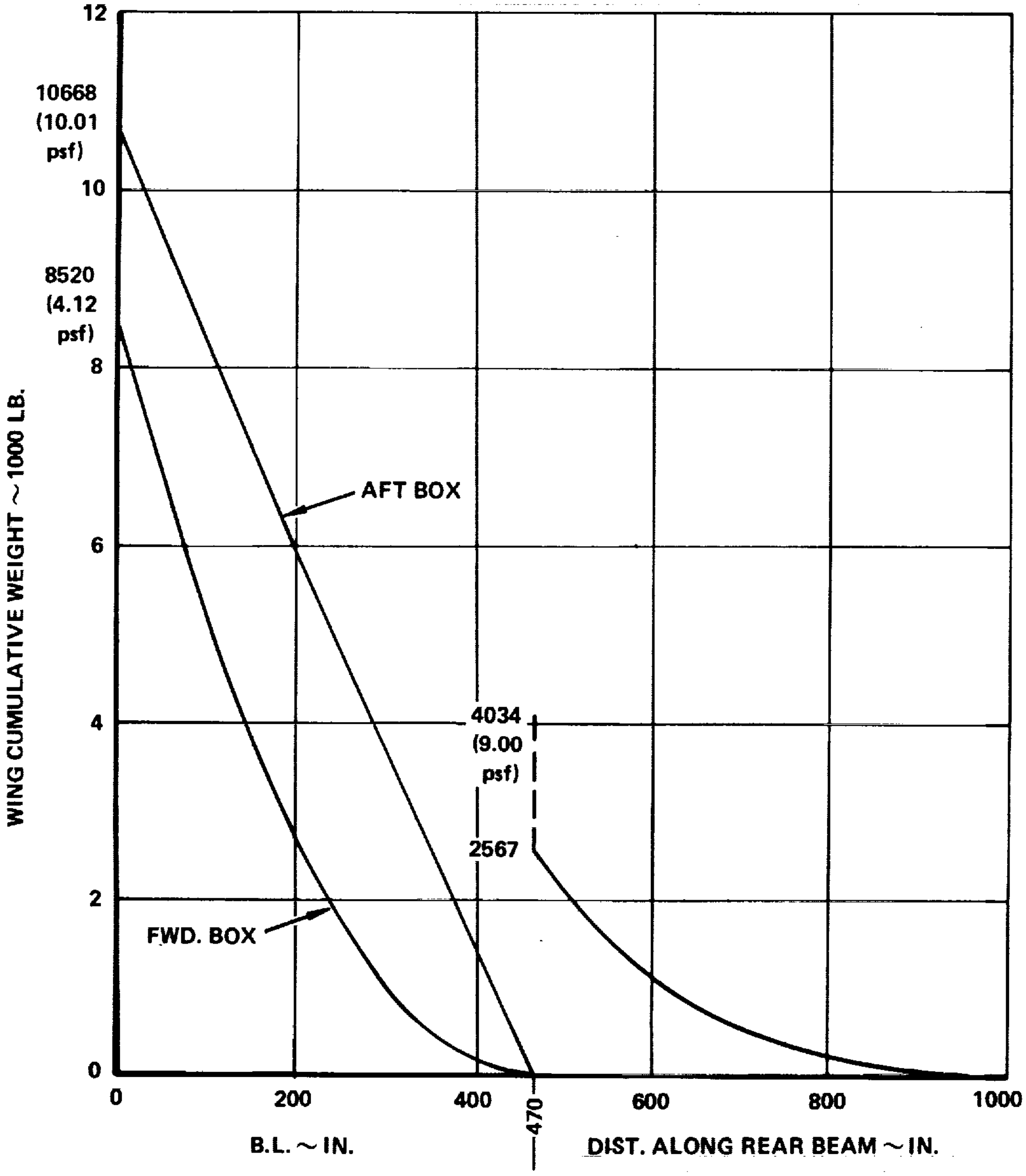
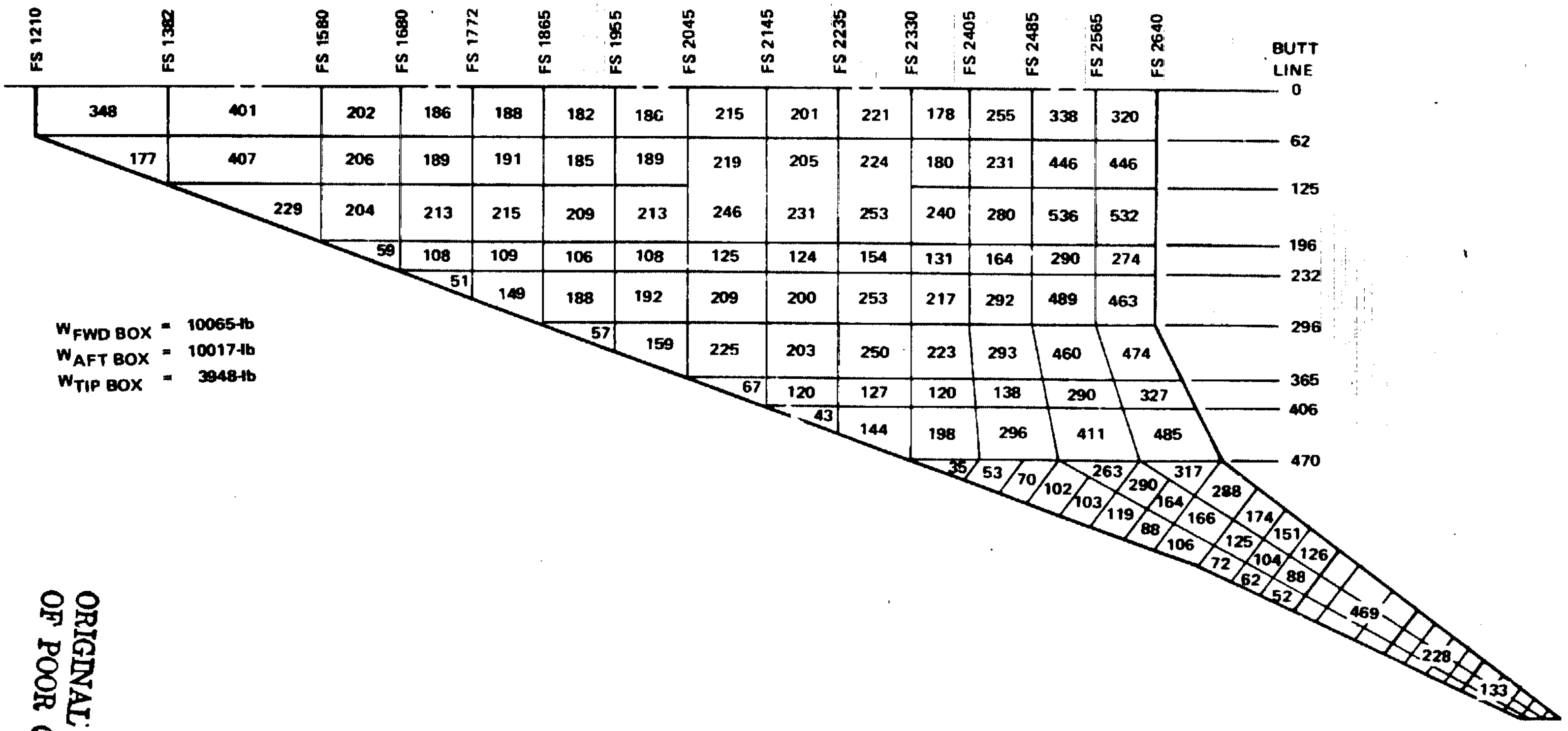


Figure 15-15. Wing Structure Mass Estimate for Chordwise Stiffened Design

TABLE 15-27. PANEL LOAD AND UNIT WEIGHT - SPANWISE STIFFENED DESIGN

PANEL NO.	INPLANE LOADS N_X N_Y N_{XY} (lb/in.)	PRESSURE LOADS UPR. LWR. $\sum$ (lb/in^2)	PANEL UNIT WEIGHTS TOTAL UPR LWR SPARS RIBS (lb/ft^2)	PANEL ESTIMATED WEIGHTS W_{PANEL} (A) (lb/ft^2)
40322	11 −1,185 −290	−8.33 −9.47 17.80	4.70 1.50 1.20 0.80 1.20	3.81
40236	306 −16,986 −2,542	−8.96 −10.14 19.10	12.75 4.80 5.20 1.40 1.35	13.84
40536	518 −16,409 −4,174	−7.47 −8.29 15.76	13.75 5.20 6.35 1.10 1.10	14.66
41036	−450 −9,499 −3,227	1.27 0.11 1.33	9.25 3.70 4.20 0.75 0.60	8.04
41316	163 −17,949 4,292	4.98 0.26 5.24	13.78 5.70 6.35 1.13 0.60	13.53
41348	−1,028 −9,412 −2,750	5.07 1.0 6.07	8.50 3.40 3.65 0.55 0.90	8.60

(A) $W_{PANEL} = \left[|N_X| + |N_Y| + |2N_{XY}| + 350|\Delta p| \right] \div 2100$



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FS 2565
FS 2640
BUTT LINE
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348 401 202 186 188 182 186 215 201 221 178 255 338 320
177 407 206 189 191 185 189 219 205 224 180 231 446 446
229 204 213 215 209 213 246 231 253 240 280 536 532
59 108 109 106 108 125 124 154 131 164 290 274
51 149 188 192 209 200 253 217 292 489 463
57 159 225 203 250 223 293 460 474
67 120 127 120 138 290 327
43 144 198 296 411 485
35 53 70 102 263 317 288
103 290 174
119 164 166 151
88 106 125 104 126
72 62 88 52
469
228
133
$W_{FWD\ BOX}$ = 10065-lb
$W_{AFT\ BOX}$ = 10017-lb
$W_{TIP\ BOX}$ = 3948-lb

Figure 15-16. Optimum Panel Weights (lb/side) - Spanwise Stiffened Design

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A build-up of the total weight for each box segment is tabulated on Figure 15-17. Cumulative plots of the strength-design weights versus span are also shown.

Monocoque Wing Design

This section presents the initial screening and detailed concept analysis mass estimation results for the monocoque designs. The three surface panel concepts investigated are as follows:

- Honeycomb sandwich - aluminum brazed - welded closures
- Honeycomb sandwich - aluminum brazed - mechanical fasteners
- Truss-core sandwich - mechanical fasteners

The relative weights for these 3 concepts are shown in Figure 15-18. These initial weights are for a 20-inch spar spacing and do not include allowance for weight increments for fail-safe design or flutter suppression. The data shown on Table 15-28 includes allowance for fail-safe and flutter suppression requirements. Appropriate spar spacing, as shown on the structural arrangement drawings of Section 18, are also considered in determining the detailed weights presented.

The basic unit weight data resulting from the detailed concept analysis are presented in Section 12 for the monocoque designs. These data identify unit weight of the surface panels, substructure, and combined surface panels and substructure at each point design region for strength requirements.

The scaling equation used for weight interpolation between the design analysis panels is shown in Table 15-29 for the welded closure concept:

$$w\ (\text{lb/ft}^2) = \left[\left(\left|N_x\right| + \left|N_y\right| + \left|2N_{xy}\right| + 350\left|\Delta P\right|\right) \div (3000)\right]$$

with the notation as described earlier for the Chordwise Stiffened Design Concepts. Panel weights resulting from this interpolation process are shown in Figure 15-19 for the welded closure design.

Weight equations were developed for the wing forward, aft and tip box areas which depend upon the detail stress analysis at each point design region:

SPANWISE STIFFENED - HAT SECTION

	WEIGHT/SIDE (LB.)		
	FWD. BOX	AFT BOX	TIP BOX
STRENGTH ONLY	10,065	10,017	3,948
NON-OPTIMUM	2,617	2,604	1,026
FAIL-SAFE	—	—	—
FLUTTER			1,464
TOTAL	12,682	12,621	6,438

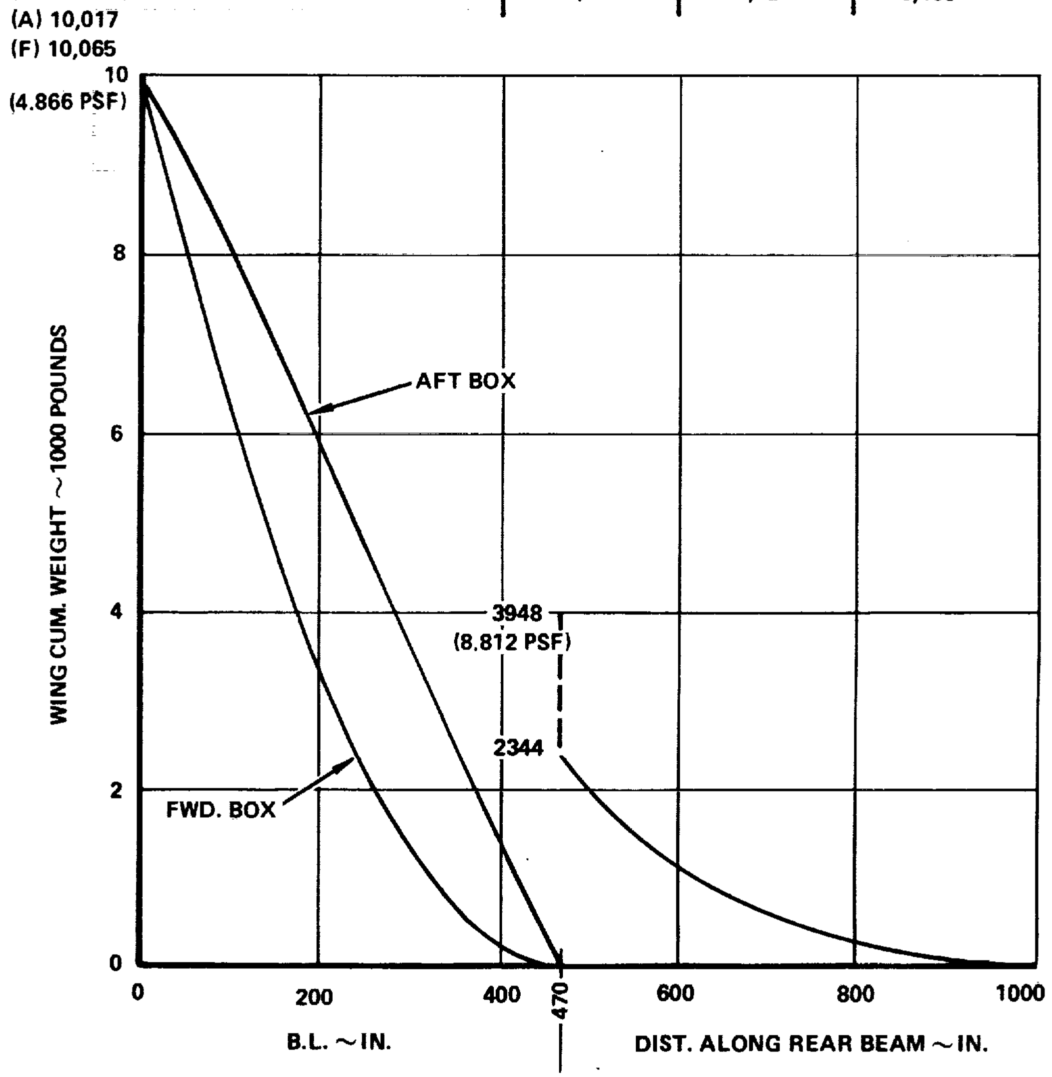

Figure 15-17. Wing Structure Mass Estimate for Spanwise Stiffened Design

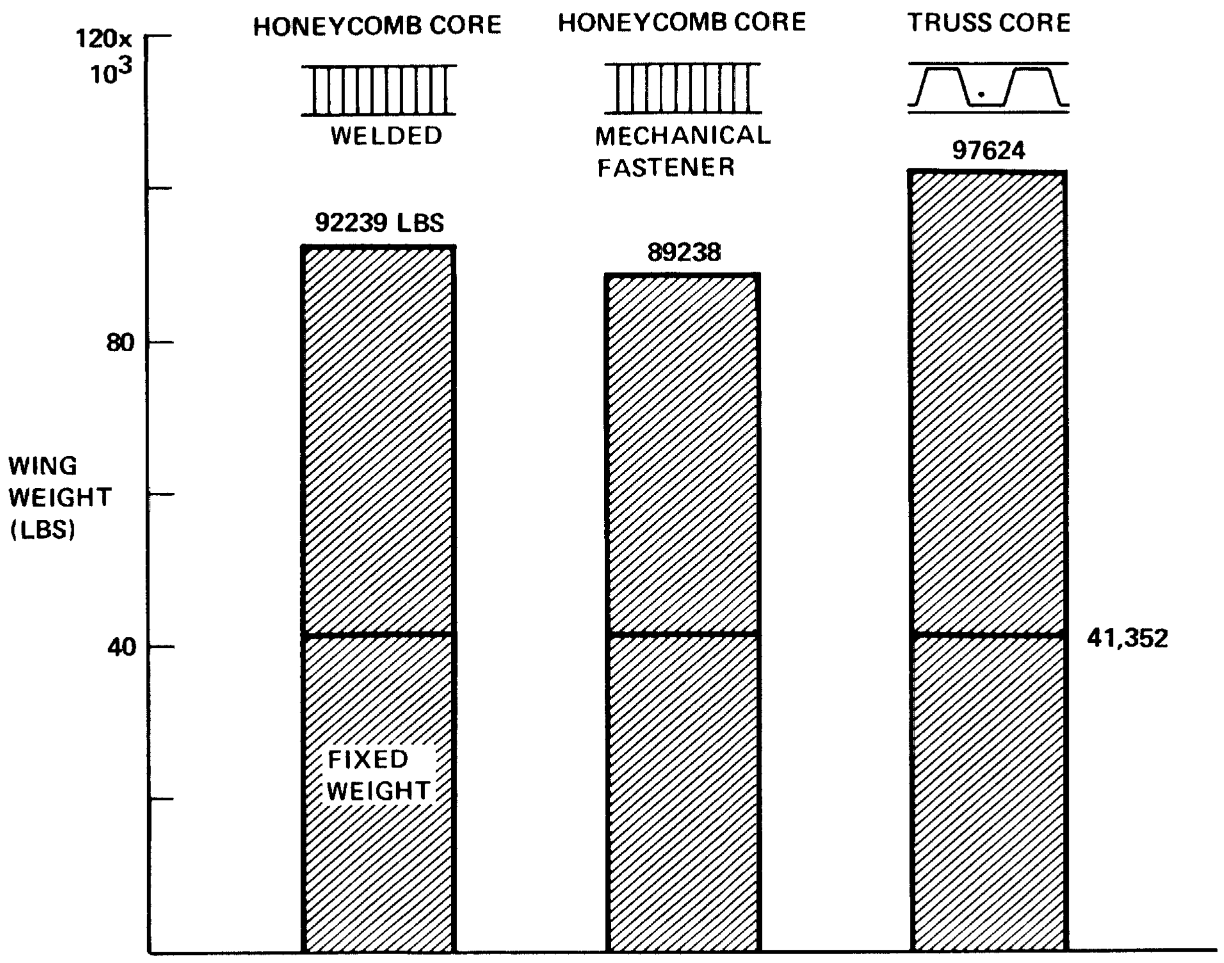

Figure 15-18. Concept Weight Comparison - Monocoque Arrangement

TABLE 15-28. ESTIMATED WING MASS - MONOCOQUE DESIGN CONCEPT

ITEM		HONEYCOMB MECH. FAST.	HONEYCOMB WELDED	TRUSSCORE
VARIABLE WEIGHT	(lb)	50,978	53,794	59,066
FORWARD BOX	(lb)	(21,982)	(24,057)	(28,667)
SURFACES		14,656	14,386	20,104
SPARS		4,616	5,965	5,502
RIBS		2,710	3,706	3,061
AFT BOX	(lb)	(19,692)	(20,153)	(20,748)
SURFACES		13,984	13,717	14,948
SPARS		4,060	4,523	3,945
RIBS		1,648	1,913	1,855
TIP BOX	(lb)	(9,304)	(9,584)	(9,651)
SURFACES		8,059	8,059	8,424
SPARS		928	1,044	806
RIBS		317	481	421
FIXED WEIGHT	(lb)	41,352	41,352	41,352
TOTAL WING WEIGHT	(lb)	92,330(A)(D)	95,146(B)(D)	100,418(C)(D)

(A) INCLUDES A FAIL-SAFE WEIGHT INCREMENT OF 752 LBS.

(B) INCLUDES A FAIL-SAFE WEIGHT INCREMENT OF 567 LBS.

(C) INCLUDES A FAIL-SAFE WEIGHT INCREMENT OF 454 LBS.

(D) INCLUDES A FLUTTER WEIGHT INCREMENT OF 2,340 LBS.

TABLE 15-29. PANEL LOAD AND UNIT WEIGHT - MONOCOQUE WELDED DESIGN

PANEL NO.	INPLANE LOADS N_X N_Y N_{XY} (lb/in.)	PRESSURE LOADS UPR. LWR. $\sum$ (lb/in^2)	PANEL UNIT WEIGHTS TOTAL UPR LWR SPARS RIBS (lb/ft^2)	PANEL ESTIMATED WEIGHT $W_{PANEL}^{(A)}$ (lb/ft^2)
40322	51 −529 −191	−8.33 −9.47 17.80	4.47 1.14 0.98 1.65 0.70	2.40
40236	−1,193 −11,638 −2,099	−8.96 −10.14 19.10	8.60 2.51 2.91 2.22 0.96	7.91
40536	−3,272 −11,787 −4,795	−7.47 −8.29 15.76	8.64 2.92 3.30 1.72 0.70	10.05
41036	−2,219 −6,423 −3,209	1.27 0.11 1.33	5.37 1.87 1.94 1.09 0.47	5.18
41316	−1,587 −12,183 +3,310	4.98 0.26 5.24	7.25 2.71 3.14 1.01 0.39	7.40
41348	−1,190 −7,263 +3,285	5.07 1.0 6.07	5.70 2.02 2.24 1.12 0.33	5.72

(A) $W_{PANEL} = \left[|N_X| + |N_Y| + |2N_{XY}| + 350|\Delta p| \right] \div 3000$

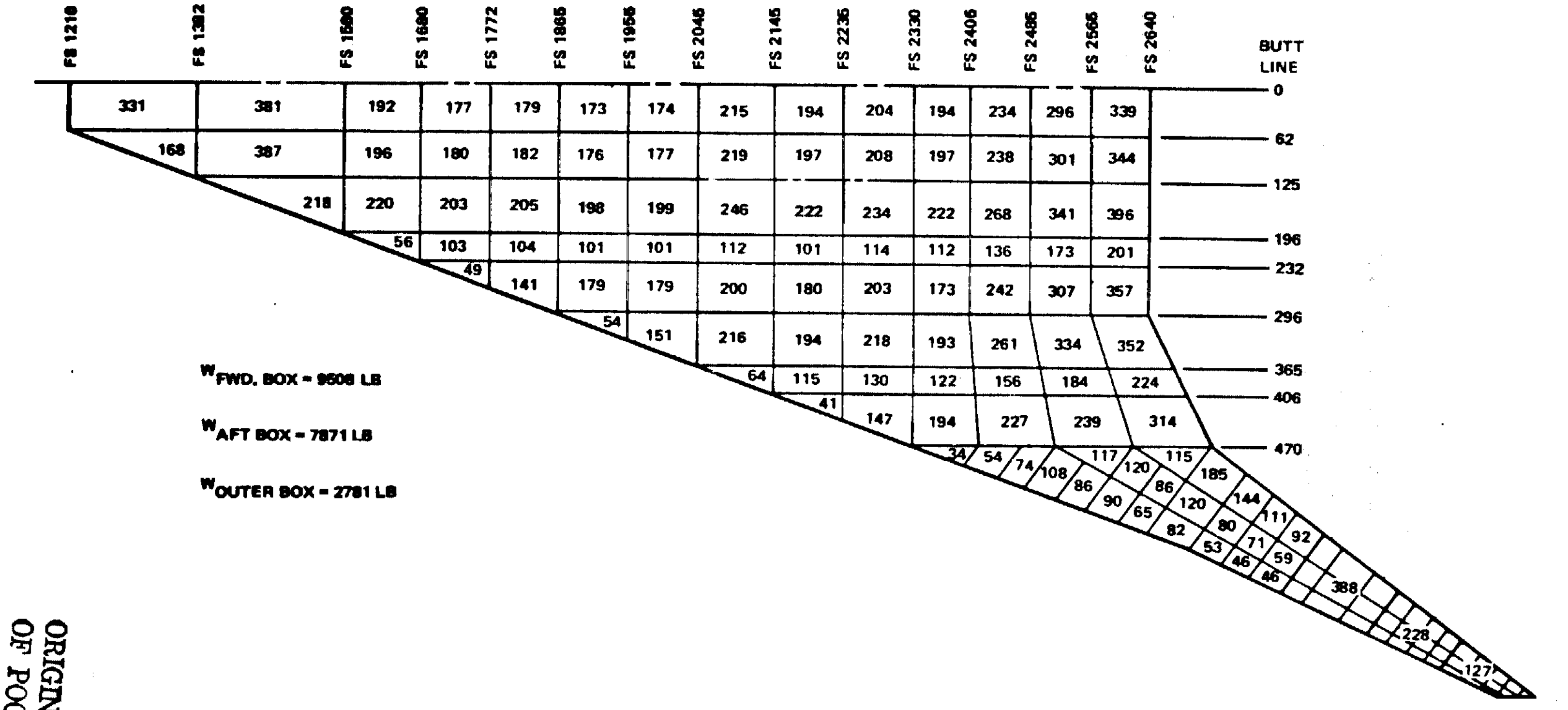
FS 1218
FS 1382
FS 1580
FS 1680
FS 1772
FS 1865
FS 1956
FS 2046
FS 2145
FS 2236
FS 2330
FS 2406
FS 2486
FS 2566
FS 2640
BUTT LINE
0
62
125
196
232
296
365
406
470
331 381 192 177 179 173 174 215 194 204 194 234 296 339
168 387 196 180 182 176 177 219 197 208 197 238 301 344
218 220 203 205 198 199 246 222 234 222 268 341 396
56 103 104 101 101 112 101 114 112 136 173 201
49 141 179 179 200 180 203 173 242 307 357
54 151 216 194 218 193 261 334 352
64 115 130 122 156 184 224
41 147 194 227 239 314
34 54 74 108 117 115
86 120 185
90 86 144
65 120 111
82 80 92
53 71 59
46 46
388
228
127
$W_{FWD.\ BOX}$ = 9508 LB
$W_{AFT\ BOX}$ = 7871 LB
$W_{OUTER\ BOX}$ = 2781 LB

Figure 15-19. Optimum Panel Weights - Monocoque Design (Welded Closures)

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$$w_{fb} = k_{fb} \left(6w_{40322} + w_{41036}\right)$$

$$w_{ab} = k_{ab} \left(w_{40236} + w_{40536} + w_{41036}\right)$$

$$w_{tb} = k_{tb} \left(w_{41316} + 2w_{41348}\right)$$

Using the welded closure configuration as an example, Table 15-30 presents the method used to derive component weights shown in Table 15-31 and summarized in Table 15-28. The basic data used for the analyses are presented in Table 15-32.

The damage tolerance and flutter suppression weight increments are superimposed on the above results. A flutter penalty of 2340 pounds is identified in Figure 15-14 for the monocoque design. This value is an estimated amount over the strength-design requirements to be applied to the stiffness critical wing tip structure. Fail-safe critical areas are identified for both the wing tip (inboard) and the aft box in Figure 15-20. The weight increment is based on fail-safe analysis of the 2 point design regions indicated. The results of these analyses, as shown in Section 13, indicates that sizable penalties are required to meet the fail-safe requirements. However, the use of the fail-safe reinforcement provides additional cross sectional area which reduces the spanwise ($N_y \div 1.5\ \bar{t}$) limit stress level from 52 ksi to 35 ksi or approximately 33 percent. This permits further reduction of the surface panel thickness and fail-safe reinforcement. This load redistribution process results in a surface panel thickness and fail-safe reinforcement combination shown on Table 15-33 and 15-34 for point design regions 40536 and 41348, respectively. As indicated on the tables, these results are applied to establish the mass increment to satisfy the fail-safe requirements for the wing aft box and tip box structure. The applicable areas (Figure 15-20) were obtained by reviewing the critical inplane loads and surface panel thicknesses and comparing the resulting limit stresses to the stress levels at the respective point design regions. The tables further define the weight increment for the various insert/closures used with the honeycomb panel design. It is noted that the welded closure method requires a smaller fail-safe increment than the mechanically fastened approach. Also, the welded method results in 540 pound reduction in fuel tank sealant in the forward and aft boxes. Unfortunately, the panel edges required for sufficient weld thickness and module-approach of assembly results in the welded design to be three-percent heavier than the mechanically fastened design.

TABLE 15-30. COMPONENT UNIT WEIGHT DERVIATION - MONOCOQUE WELDED DESIGN

OPTIMUM BOX WEIGHT (lb/side)		÷	PLANFORM AREA (ft^2/side)	=	OPTIMUM UNIT WT. (psf)	×	NON OPT FACTOR	=	UNIT WT. (INCL N.O.F.) (psf)
FWD	9508		2068.3		4.597		1.26		5.792
AFT	7871		1066.2		7.382		1.26		9.302
TIP	2781		448.0		6.207		1.26		7.821

UNIT WEIGHT EQUATION (20-inch SPAR SPACING):

$$w_{fb} = 0.180\,(6 \times 4.47 + 5.37) = 5.792 \text{ psf}$$

$$w_{ab} = 0.4114\,(8.6 + 8.64 + 5.37) = 9.302$$

$$w_{ob} = 0.4194\,(7.25 + 2 \times 5.70) = 7.821$$

TABLE 15-31. ESTIMATED WING STRUCTURE MASS - MONOCOQUE (WELDED) CONCEPT

ITEM		UNIT WEIGHT (psf)	BOX WEIGHT (lb)
FWD. BOX (AREA = 4136.6 ft^2/AIRCRAFT)	TOTAL =	5.816	24,057
SURFACES	w = 0.180 (6 X 2.63 + 3.9) =	3.543	14,656
	REDUCED TANK SEALANT REQUIREMENT =	-0.065	-270
SPARS	w = 0.180 (6 X 1.16 + 1.05) =	1.442	5,965
RIBS	w = 0.180 (6 X 0.75 + 0.48) =	0.896	3,706
AFT BOX (AREA = 2132.4 ft^2/AIRCRAFT)	TOTAL =	9.451	20,153
SURFACES	REDUCED TANK SEALANT REQUIREMENT =	-0.125	-267
	w = 0.4114 (5.54 + 6.5 + 3.9) =	6.558	13,984
SPARS	w = 0.4114 (2.08 + 1.54 + 1.05) =	1.921	4,523
	FAIL-SAFE PROVISIONS =	0.20	
RIBS	w = 0.4114 (1.00 + 0.70 + 0.48) =	0.897	1,913
OUTER BOX (AREA = 896 ft^2/AIRCRAFT)	TOTAL =	10.697	9,584
SURFACES	w = 0.4194 (6.22 + 2 X 4.50) =	6.383	5,719
	FLUTTER PREVENTION =	2.612	+2,340
SPARS	w = 0.4194 (0.72 + 2 X 0.85) =	1.015	1,044
	FAIL-SAFE PROVISIONS =	0.150	
RIBS	w = 0.4194 (0.48 + 2 X 0.40) =	0.537	481

TABLE 15-32. BASIC UNIT WEIGHT DATA FOR MONOCOQUE CONCEPTS

DESIGN CONCEPT	POINT DESIGN	SPAR SPAC. (in.)	UNIT WEIGHTS (psf)			
			SURFACES	SPARS	RIBS	TOTAL
HONEYCOMB SANDWICH BRAZED-WELDED	40322	34.0	2.63	1.16	0.75	4.54
	40236	23.4	5.54	2.08	1.00	8.62
	40536	23.4	6.50	1.54	0.70	8.74
	41036	23.4	3.90	1.05	0.48	5.43
	41316	35.0	6.22	0.72	0.48	7.42
	41348	30.0	4.50	0.85	0.40	5.75
HONEYCOMB SANDWICH BRAZED-MECH. FASTENERS	40322	34.0	2.63	0.86	0.54	4.03
	40236	23.4	5.54	1.85	0.88	8.27
	40536	23.4	6.50	1.35	0.60	8.45
	41036	23.4	3.90	0.82	0.40	5.12
	41316	35.0	6.22	0.56	0.34	7.12
	41348	30.0	4.50	0.74	0.26	5.50
TRUSSCORE SANDWICH MECH. FASTENERS	40322	34.0	3.75	1.10	0.60	5.45
	40236	23.4	5.85	1.90	0.90	8.65
	40536	23.4	6.70	1.35	0.70	8.75
	41036	23.4	4.50	0.80	0.50	5.80
	41316	35.0	6.50	0.50	0.50	7.50
	41348	30.0	4.85	0.60	0.30	5.75

REFER TO SECTION 12 STRUCTURAL CONCEPT ANALYSIS

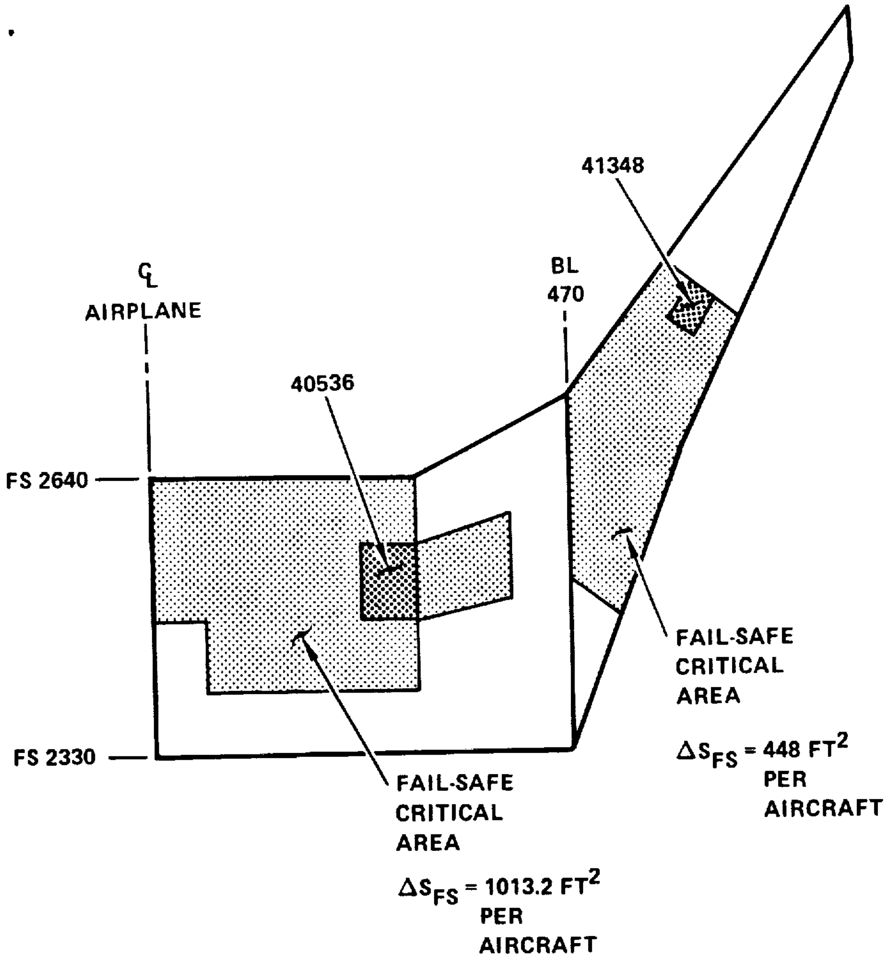
41348
℄
AIRPLANE
BL
470
40536
FS 2640
FS 2330
FAIL-SAFE
CRITICAL
AREA
ΔS_{FS} = 448 FT^2
PER
AIRCRAFT
FAIL-SAFE
CRITICAL
AREA
ΔS_{FS} = 1013.2 FT^2
PER
AIRCRAFT

Figure 15-20. Fail-Safe Critical Areas of Wing Structure

TABLE 15-33. FAIL-SAFE WEIGHT INCREMENT - AFT BOX

HONEYCOMB SANDWICH PANEL DESIGN CONCEPT	SPAR SPAC.	MIN. DESIGN	SKIN THICK.	MFG. JOINT	FAIL-SAFE REQUIREMENT		NO. STRAP	UNIT WEIGHT	DELTA WEIGHT	EFF. PLAN. AREA	DELTA WEIGHT
POINT DESIGN 40536	b (in.)	A_{SPAR} (in^2)	$\bar{t}_s$ (in.)	A_{WELD} (in^2)	A_{SPAR} (in^2)	A_{STRAP} (in^2)	n –	w (lb/ft^2)	Δw (lb/ft^2)	ΔS (ft^2)	ΔW (lb)
METALLIC INSERT – FASTENERS											
STRENGTH – DESIGN	20	0.181	0.137	–	–	–	–	3.36	–	1031	–
FAIL-SAFE – NO REDISTRIBUTION	20	–	0.137	–	0.862	0.733	1	4.49	1.63	1031	1680
FAIL-SAFE – REDISTRIBUTION	20	–	0.115	–	0.583	0.496	1	3.89	0.53	1031	546
METALLIC INSERT – WELDED											
STRENGTH – DESIGN	20	0.181	0.137	0.181	–	–	–	3.57	–	1031	–
FAIL-SAFE – NO REDISTRIBUTION	20	–	0.137	0.862	0.862	–	–	5.14	1.57	1031	1619
FAIL-SAFE – REDISTRIBUTION	20	–	0.115	0.583	0.583	–	–	3.99	0.42	1031	433
DENSIFIED CORE – FASTENERS											
STRENGTH – DESIGN	20	0.088	0.137	–	–	–	–	3.26	–	1031	–
FAIL-SAFE – NO REDISTRIBUTION	20	–	0.137	–	0.733	0.733	1	4.84	1.58	1031	1628
FAIL-SAFE – REDISTRIBUTION	20	–	0.115	–	0.496	0.496	1	3.79	0.53	1031	546

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TABLE 15-34. FAIL-SAFE WEIGHT INCREMENT - TIP BOX

HONEYCOMB SANDWICH PANEL DESIGN CONCEPT	SPAR SPAC.	MIN. DESIGN	SKIN THICK.	MFG. JOINT	FAIL-SAFE REQUIREMENT		NO. STRAP	UNIT WEIGHT	DELTA WEIGHT	EFF. PLAN. AREA	DELTA WEIGHT
POINT DESIGN 41348	b (in.)	A_{SPAR} (in^2)	t_s (in.)	A_{WELD} (in^2)	A_{SPAR} (in^2)	A_{STRAP} (in^2)	n –	w (lb/ft^2)	Δ w (lb/ft^2)	ΔS (ft^2)	ΔW (lb)
METALLIC INSERT – FASTENERS											
STRENGTH – DESIGN	20	0.181	0.091	–	–	–	–	2.30	–	–	–
FAIL-SAFE – NO REDISTRIBUTION	20	–	0.091	–	0.68	0.58	1	3.54	1.24	448	555
FAIL-SAFE – REDISTRIBUTION	20	–	0.076	–	0.46	0.39	1	2.73	0.43	448	193
METALLIC INSERT – WELDED											
STRENGTH – DESIGN	20	0.181	0.091	0.181	–	–	–	2.51	–	–	–
FAIL-SAFE – NO REDISTRIBUTION	20	–	0.091	0.68	0.68	–	–	3.66	1.15	448	515
FAIL-SAFE – REDISTRIBUTION	20	–	0.076	0.46	0.46	–	–	2.81	0.30	448	134
DENSIFIED CORE – FASTENERS											
STRENGTH – DESIGN	20	0.088	0.091	–	–	–	–	2.19	–	–	–
FAIL-SAFE – NO REDISTRIBUTION	20	–	0.091	–	0.58	0.58	1	3.42	1.23	448	551
FAIL-SAFE – REDISTRIBUTION	20	–	0.076	–	0.39	0.39	1	2.65	0.46	448	206

Composite Reinforced-Chordwise Stiffened Wing Design

The chordwise stiffened arrangement described earlier, provides the basic approach offering the maximum mass savings potential for application of composites to the design. The two composite reinforced designs investigated are as follows:

- Composite reinforced spar caps with metallic beaded surface panels
- Composite reinforced spar caps and surface panels

A comparison of these two reinforcing methods are presented in Table 15-35 with their all-metallic counterpart. The results show an 11-percent to 13-percent reduction in total wing weight. Initial screening data used to derive this comparison is reported in Section 12.

Since the aforementioned results indicated that both concept weights were very close to each other, further detailed analyses were conducted. The results of the latter is presented in Tables 15-36 and 15-37. An interesting conclusion, when comparing the reinforced spar caps only with the all metallic design, is a one pound reduction in structure weight for each 0.40-pound of B/PI composite reinforcing material used.

Using the same equations as described earlier in the Chordwise Stiffened Design section, the box component weights are derived in Tables 15-38 and 15-39. Figure 15-21 shows the relationship between the all-metallic and composite reinforced spar cap optimum panel unit weights. For a minimum-gage all-metallic design (3.8 psf) there is no weight reduction possible by reinforcing the spar caps, since no further reduction in gages is possible. However, for a highly loaded all-metallic panel weighing 15 pounds per square foot, the addition of 2.4 pounds per square foot of composite reinforcement to the spar caps will reduce the overall panel weight to 9 pounds per square foot.

Wing Tip Mass for Structural Arrangements

Surface panel shear thickness is a critical parameter for evaluation of outer wing torsional stiffness and flutter speed. Therefore, the estimated thicknesses used in the NASTRAN 2-D model were compared with those derived from the detailed stress analysis (strength design only) as shown in Figures 15-22 through 15-24. This data

TABLE 15-35. COMPARISON OF COMPOSITE REINFORCED DESIGNS

ARRANGEMENT	CHORDWISE STIFFENED		
PANEL CONCEPT	CONVEX BEADED		
MATERIAL APPLICATION	ALL METALLIC	COMPOSITE REINFORCED	
		SURFACE AND SPAR CAPS	SPAR CAPS
(SEE NOTES)	(A)	(B)	(A)
VARIABLE WEIGHT	(56,655)	(44,474)	(43,624)
CENTER SECTION	(8,722)	(6,847)	(6,716)
UPPER SURFACE	1,570	1,689	1,570
LOWER SURFACE	1,570	1,680	1,570
SPAR CAPS AND WEBS	4,884	2,780	2,878
RIBS	698	698	698
INTERM PANEL	(39,296)	(30,847)	(30,258)
UPPER SURFACE	7,073	7,610	7,073
LOWER SURFACE	7,073	7,593	7,073
SPAR CAPS AND WEBS	22,006	12,500	12,963
RIBS	3,144	3,144	3,144
OUTER PANEL	(8,637)	(6,780)	(6,650)
UPPER SURFACE	1,555	1,680	1,555
LOWER SURFACE	1,555	1,660	1,555
SPAR CAPS AND WEBS	4,837	2,750	2,850
RIBS	690	690	690
FIXED WEIGHT	(41,352)	(41,352)	(41,352)
TOTAL (STRENGTH DESIGN ONLY)	98,007	85,826$^{(C)}$	84,976

NOTES:

(A) 20-inch SPAR SPACING, 60-inch RIB SPACING

(B) 40-inch SPAR SPACING IN HIGHLY LOADED AREAS ONLY

(C) FORWARD BOX (FWD OF F.S. 2330) IS ALL METALLIC.
TOTAL WEIGHT IS 89,494 lbs FOR 40-inch SPAR SPACING AND
REINFORCED SURFACE AND CAPS THROUGHOUT.

TABLE 15-36. DETAILED WEIGHTS FOR COMPOSITE REINFORCED DESIGNS - SURFACE AND SPAR CAP

A. COMPOSITE REINFORCED (B/PI) SURFACE AND SPAR CAPS – 40 inch SPAR SPACING

POINT DESIGN REGION		COMPOSITE REINFORCED DESIGN		ALL METAL-CHORDWISE STIFFENED DESIGN		TOTAL		RATIO OF: COMPOSITE / METAL
		SURFACE	SPAR CAP	SURFACE	SPAR CAP	COMPOSITE REINFORCED DESIGN	ALL METAL DESIGN	
		psf ~ pounds per square foot						percent
40322	(U)	1.341	0.177	1.62	0.60	4.46	4.87	91.6
	(L)	1.497	0.203	1.41				
	(C)	(0.37)	(0.10)	–	–	(0.47)	–	
40236	(U)	1.464	1.071	1.63	8.30	7.73	13.48	57.3
	(L)	1.829	1.695	1.88				
	(C)	(0.58)	(2.25)	–	–	(2.83)		
40536	(U)	1.958	0.98	2.241	7.07	7.55	12.85	58.8
	(L)	1.536	1.48	1.943				
	(C)	(0.53)	(1.94)	–	–	(2.47)		
41036	(U)	1.900	0.751	1.45	5.10	6.14	9.45	65.0
	(L)	1.549	1.074	2.03				
	(C)	(0.49)	(1.30)	–	–	(1.79)		
41316	(U)	2.305	1.381	2.80	11.05	8.95	17.16	53.2
	(L)	2.120	2.201	2.37				
	(C)	(0.44)	(3.08)	–	–	(3.52)		
41348	(U)	1.978	0.936	2.20	6.86	6.56	11.38	57.6
	(L)	1.539	1.407	1.62				
	(C)	(0.53)	(1.83)	–	–	(2.36)		

NOTES: (U) = UPPER; (L) = LOWER; (C) = COMPOSITE WEIGHT INCLUDED IN UNIT WEIGHT CALCULATIONS

TABLE 15-37. DETAILED WEIGHTS FOR COMPOSITE REINFORCED DESIGNS - SPAR CAPS ONLY

B. COMPOSITE REINFORCED (B/PI) SPAR CAPS ONLY – 20 inch SPAR SPACING

POINT DESIGN REGION	SPAR CAPS			TOTAL		RATIO OF: $\frac{\text{COMPOSITE}}{\text{METAL}}$
	COMPOSITE REINFORCED		ALL-METAL DESIGN	COMPOSITE REINFORCED DESIGN	ALL-METAL DESIGN	
	COMP. ONLY	TOTAL				
	psf ~ pounds per square foot					percent
40322 (U) (L)	– –	– –	0.60	3.80	3.80	100
40236 (U) (L)	0.720 1.224	1.239 1.740	7.92	7.14	12.08	59.1
40536 (U) (L)	0.626 1.012	1.142 1.531	6.70	7.27	11.3	64.3
41036 (U) (L)	0.384 0.604	0.902 1.123	4.60	5.68	8.25	68.8
41316 (U) (L)	0.921 1.531	1.440 2.047	9.65	9.22	15.38	59.9
41348 (U) (L)	0.543 0.861	1.061 1.380	5.90	6.29	9.75	64.5

(U) = UPPER; (L) = LOWER

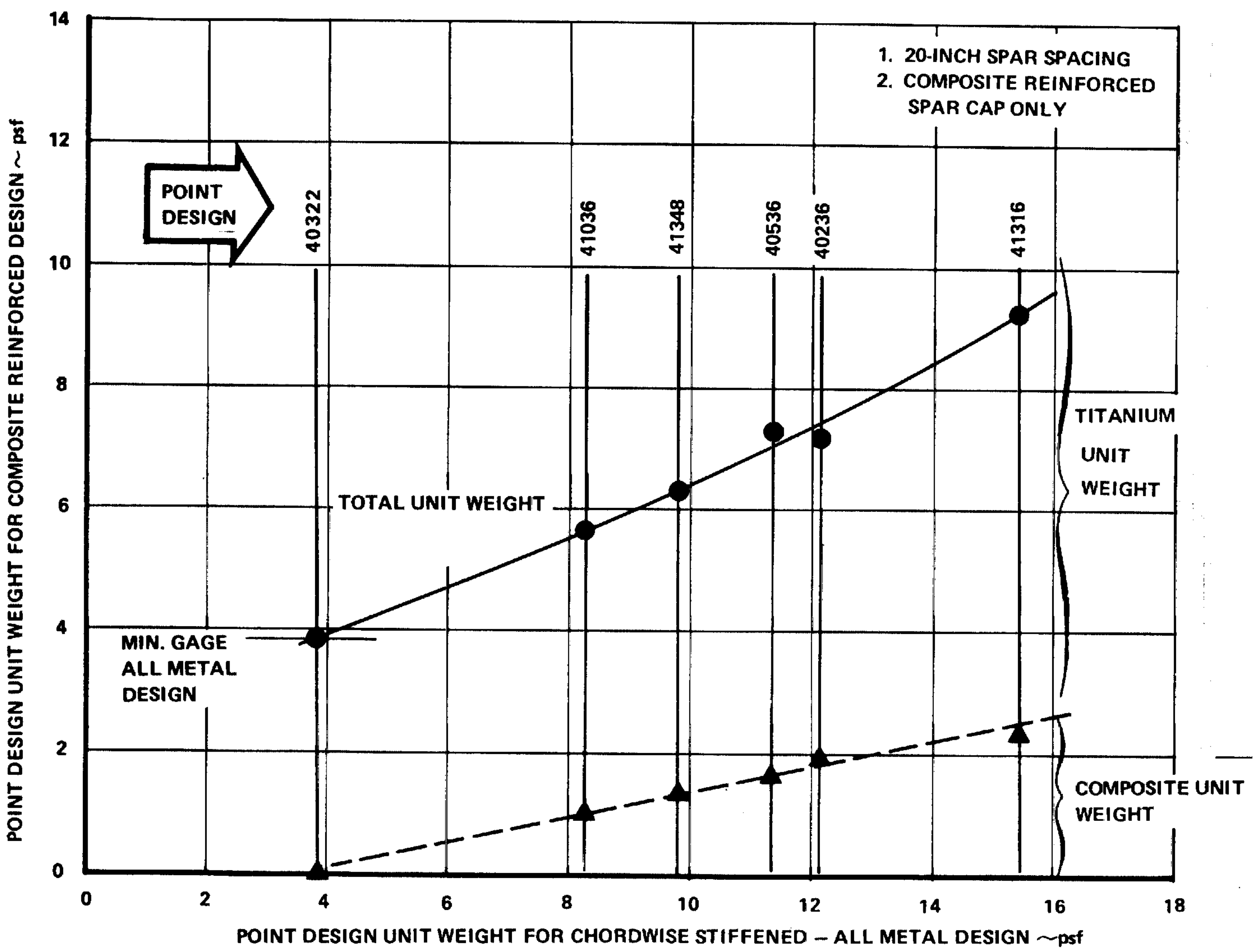


Figure 15-21 Comparison of Optimum Panel Unit Weights - Composite Reinforced Versus All-Metallic Chordwise Stiffened Designs

C.2

TABLE 15-38. COMPONENT WEIGHT DERIVATION - COMPOSITE REINFORCED SPAR CAPS ONLY

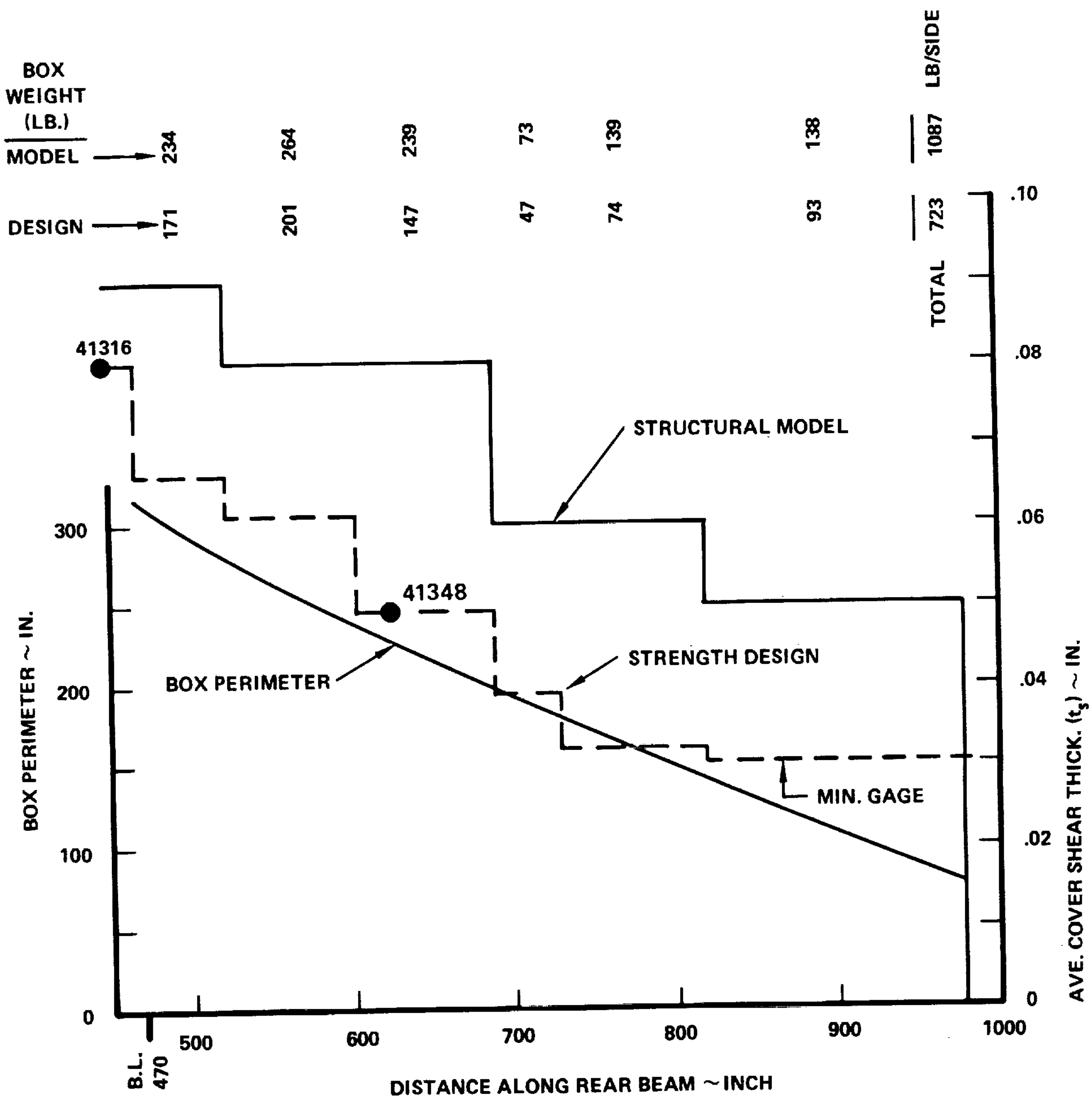
A. COMPOSITE REINFORCED SPARS ONLY (20-inch SPAR SPACING)	UNIT WEIGHT (lb/ft^2)	BOX WEIGHT (lb)
FORWARD BOX: (AREA = 4136.6 ft^2/AIRPLANE)	4.975	20,580
• SURFACES } SAME AS ALL-METALLIC (LESS FAIL-SAFE)		(9,452)
• RIBS		(2,570)
• SPARS = 0.122 (1.53 X 9 + 2.375) + FAIL-SAFE 0.090 (COMPOSITES: 522 lb)	2.069	(8,558)
AFT BOX: (AREA =2132.4 ft^2/AIRPLANE)	8.152	17,384
• SURFACES } SAME AS ALL-METALLIC LESS FAIL-SAFE)		(7,302)
• RIBS		(1,514)
• SPARS = 0.3162 (4.21 + 3.72 + 2 X 2.375) + 0.009 (COMPOSITES: 3,762 lb)	4.018	(8,568)
TIP BOX: (AREA = 896.0 ft^2/AIRPLANE)	11.292	10,118
• SURFACES } SAME AS ALL-METALLIC LESS FAIL-SAFE)		(3,459)
• RIBS		(683)
• SPARS = 0.3465 (3.97 + 2.74 + 2 X 1.53) + 0.005 = (COMPOSITES: 1,196 lb)	3.390	(3,038)
• FLUTTER INCREMENT	3.279	(2,938)

TOTAL COMPOSITE MATERIAL WEIGHT: 5,480 lb

TABLE 15-39. COMPONENT WEIGHT DERIVATION - COMPOSITE REINFORCED SPAR CAPS AND SURFACES

B. COMPOSITE REINFORCED SPARS AND SURFACES (40-INCH SPAR SPACING)	UNIT WEIGHT (lb/ft^2)	BOX WEIGHT (lb)
FORWARD BOX: (AREA = 4136.6 ft^2/AIRPLANE)	5.846	24,184
• SURFACES = 0.122 (9 X 2.838 + 3.449) + (0.006) (COMPOSITES: 1,635 lb)	(3.543)	(14,655)
• RIBS (SAME AS METALLIC)	(0.621)	(2,570)
• SPARS = 0.122 (9 X 1.31 + 2.175) – (0.022) (COMPOSITES: 919 lb)	(1.682)	(6,959)
AFT BOX: (AREA = 2132.4 ft^2/AIRPLANE)	8.719	18,592
• SURFACES = 0.3162 (3.293 + 3.49 + 2 X 3.449) = (COMPOSITES: 1,414 lb)	(4.326)	(9,225)
• RIBS (SAME AS METALLIC)	(0.710)	(1,514)
• SPARS = 0.3162 (3.996 + 3.51 + 2 X 2.175) – (0.066) = (COMPOSITES: 4,578 lb)	(3.683)	(7,853)
TIP BOX: (AREA = 896.0 ft^2/AIRPLANE)	11.954	10,711
• SURFACES = 0.3465 (4.425 + 3.517 + 2 X 2.838) = (COMPOSITES: 536 lb)	(4.719)	(4,228)
• RIBS (SAME AS METALLIC)	(0.762)	(683)
• SPARS = 0.3465 (4.062 + 2.643 + 2 X 1.31) – (0.037) (COMPOSITES: 1,586 lb)	(3.194)	(2,862)
• FLUTTER INCREMENT	(3.279)	(2,938)

TOTAL COMPOSITE MATERIAL WEIGHT: 10,668 lb.



NOTE: SHEAR THICKNESS, $(t_s) = 0.78\bar{t}$
WHERE $\bar{t}$ = EQUIVALENT WEIGHT THICKNESS

Figure 15-22. Shear Thickness of Wing Tip Structure - Chordwise Stiffened

NOTE: SHEAR THICKNESS (t_s) = 0.40$\bar{t}$

WHERE $\bar{t}$ = EQUIVALENT WEIGHT THICKNESS

BOX WEIGHT (LB.)											TOTAL LB/SIDE
MODEL →	187	124	89	92	72	51	42	44	22	60	783
DESIGN →	223	139	120	100	75	62	41	32	12	38	842

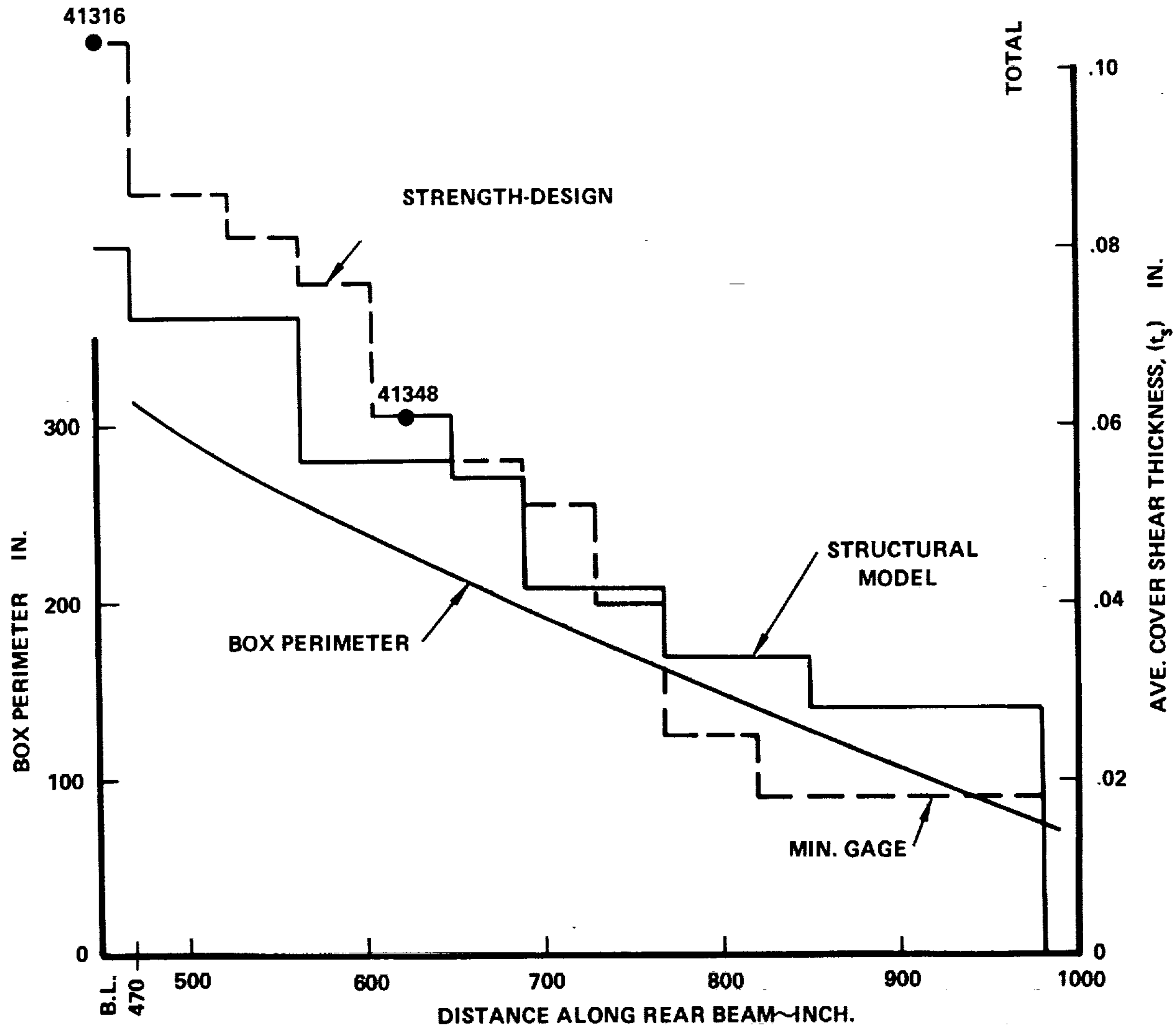


Figure 15-23. Shear Thickness of Wing Tip Structure - Spanwise Stiffened

NOTE: SHEAR THICKNESS (t_s) = 0.97$\bar{t}$
WHERE $\bar{t}$ = EQUIVALENT WEIGHT THICKNESS

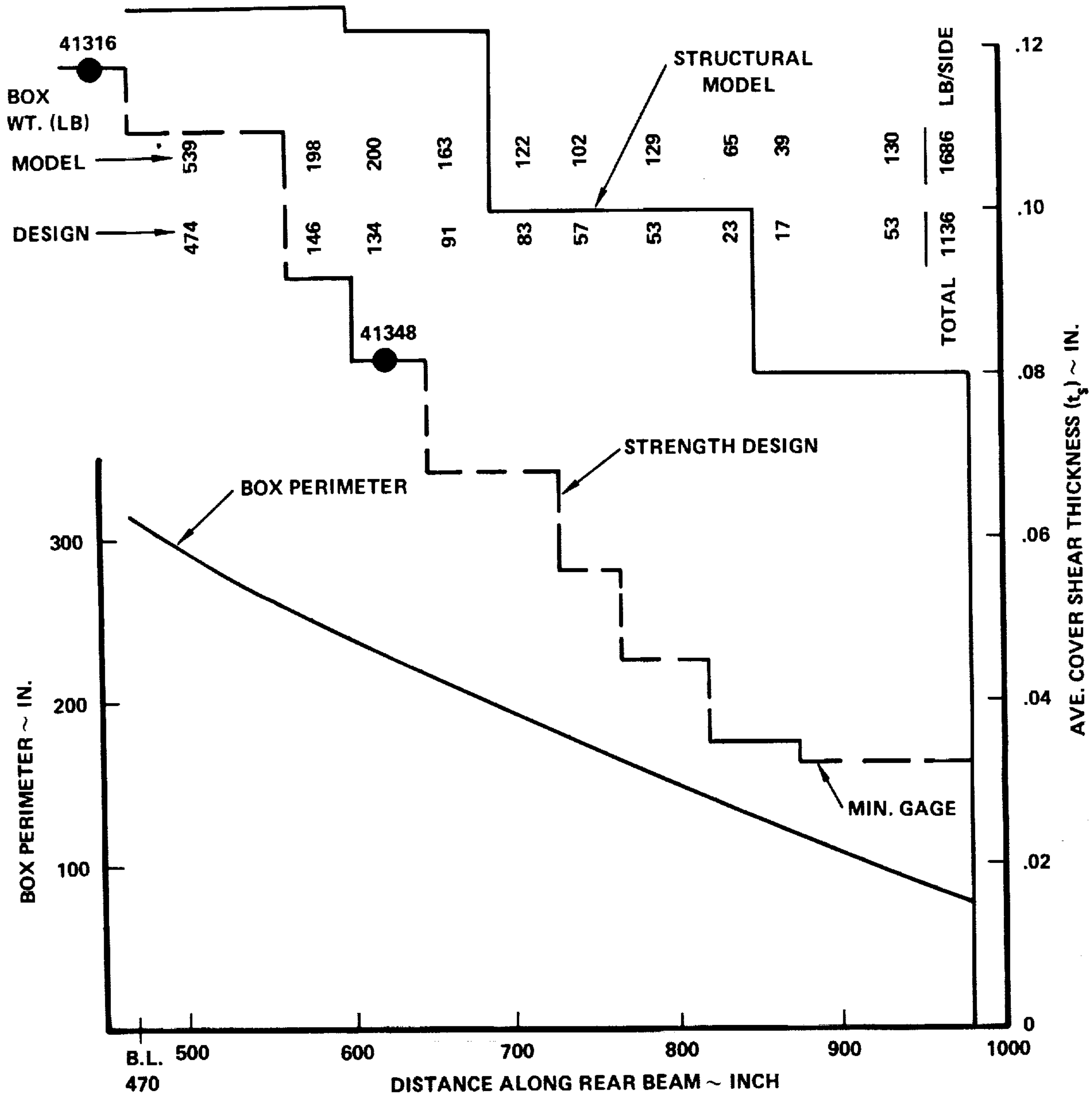


Figure 15-24. Shear Thickness of Wing Tip Structure - Monocoque

was used as the basis for evaluation of the flutter increment required for each structural concept (Reference Figure 15-14).

These mass data comparisons included only that portion of the outer box which lies perpendicular to the rear beam and outboard of butt line 470 as pictorially displayed on Figure 15-25.

From a weight efficiency standpoint, the monocoque arrangement is preferred, since 97-percent of the surface panel weight is effective in providing torsional stiffness. By comparison, the effective shear thicknesses for the chordwise and spanwise stiffened arrangements are only 78-percent and 40-percent, respectively.

Wing Tip Mass Distribution Comparison

The aeroelastic analysis of Task I used a single mass distribution for the 3 wing structural arrangements. Since wing tip mass distribution has a significant effect on flutter speed, several comparisons were made between the Task I and Task II distributions. Figure 15-26 compares the dead weight shear measured spanwise along the rear beam. The wing tip structure center of gravity at a percent of chord and wing weight distribution (pounds per inch) in the spanwise direction is compared in Figure 15-27. The wing tip box geometry is displayed in Figure 15-28 and shows that the Task II planform and wetted areas are slightly larger than Task I, while the effective span is decreased about 10 percent.

FUSELAGE STRUCTURE MASS-INITIAL SCREENING

The basic structural arrangement for the fuselage design is a uniaxial stiffened structure of skin and stringers with supporting frames. The panel structural concepts investigated are as follows:

- Zee stiffened
- Closed-hat stiffened
- Open-hat stiffened

Figure 15-29 presents the body perimeter, width, height and cross sectional areas of the fuselage. The perimeter was used to calculate the wetted areas for 200 inch

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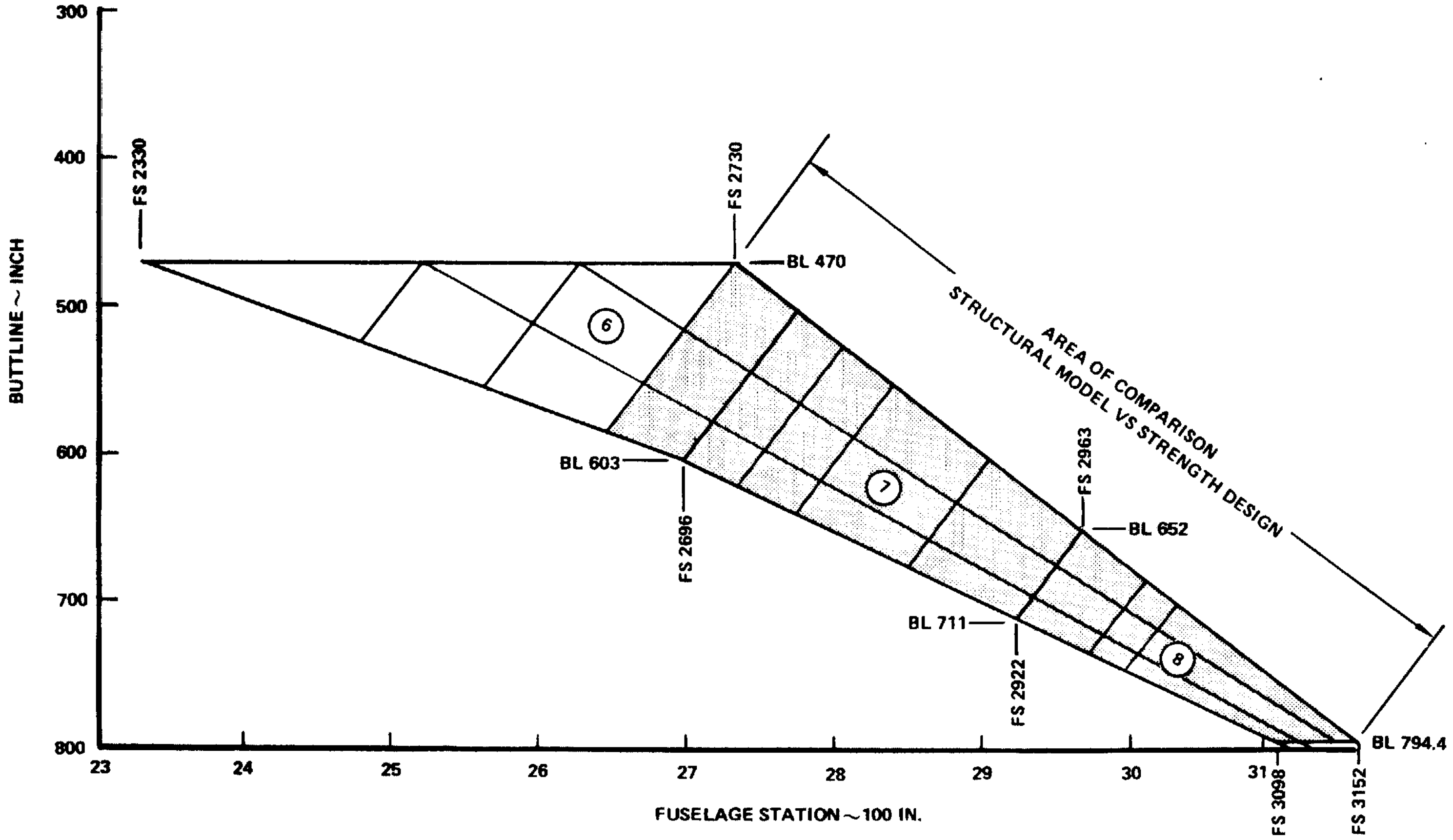

300
400
500
600
700
800
BUTTLINE ~ INCH
23
24
25
26
27
28
29
30
31
FUSELAGE STATION ~ 100 IN.
FS 2330
FS 2730
BL 470
AREA OF COMPARISON
STRUCTURAL MODEL VS STRENGTH DESIGN
6
7
8
BL 603
FS 2696
FS 2963
BL 652
BL 711
FS 2922
BL 794.4
FS 3098
FS 3152

Figure 15-25. Wing Tip Structure Planform Geometry - Task I

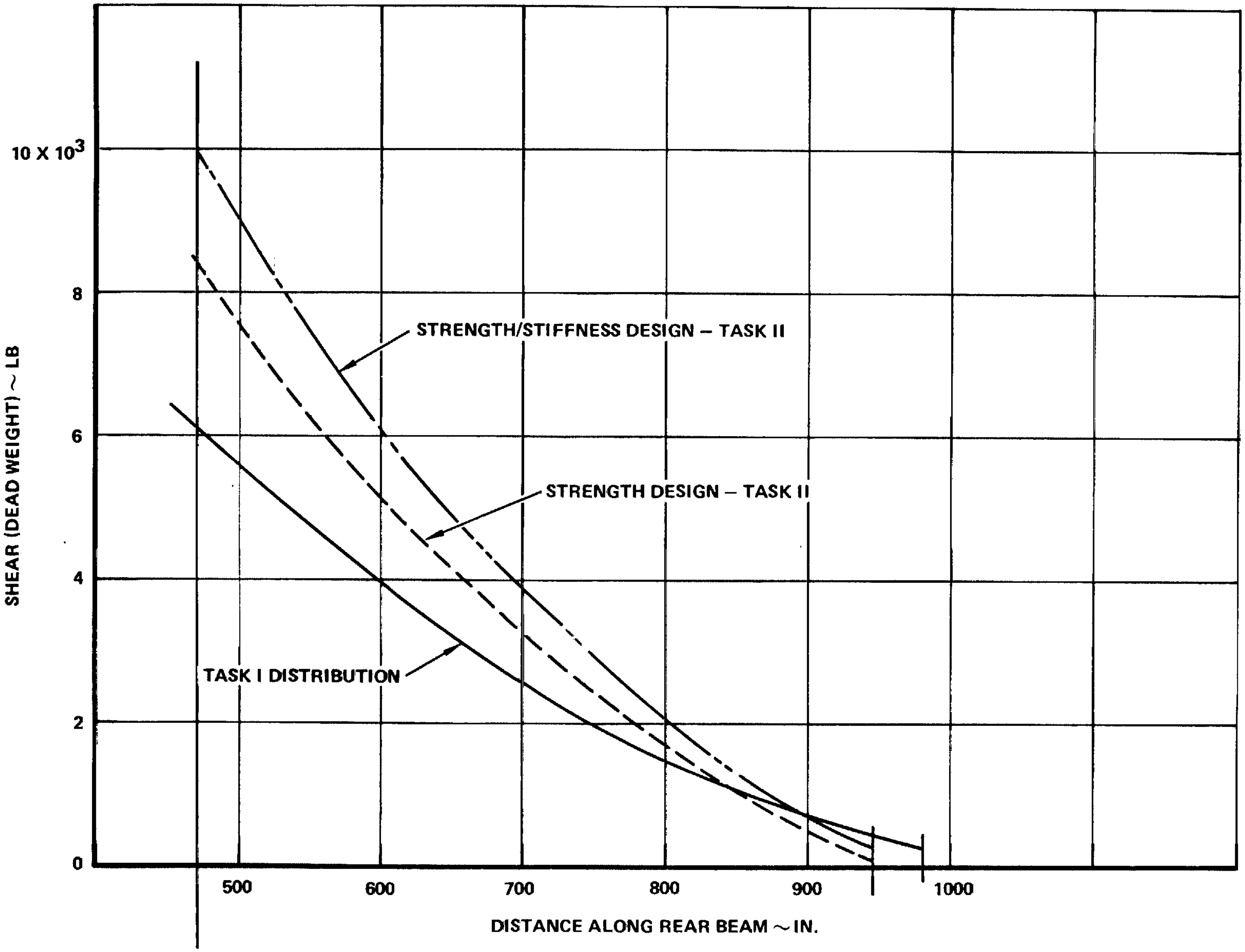

Figure 15-26 Comparison of SIC Mass Distribution - Task I and Task II

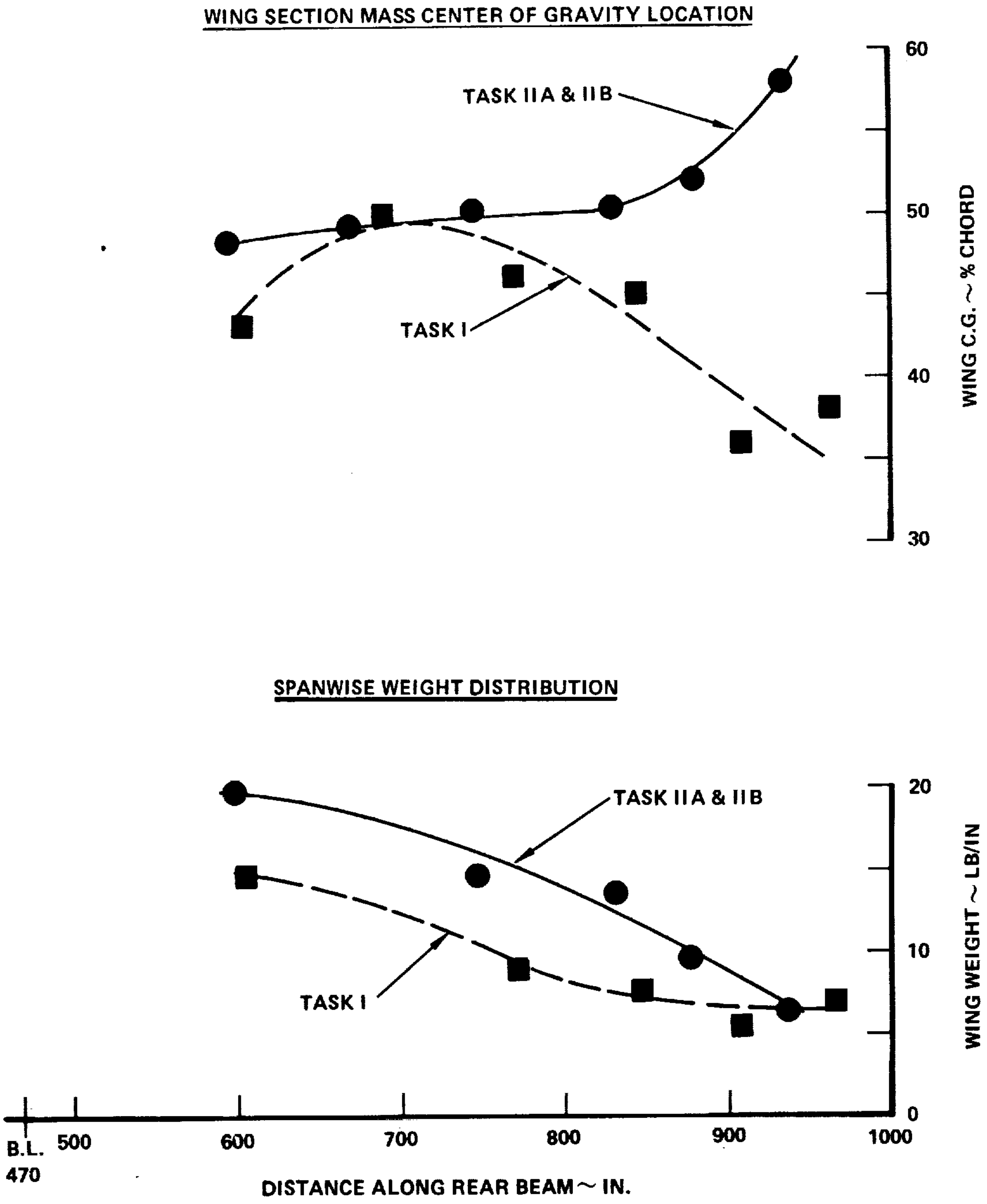

Figure 15-27. Wing Tip Mass Distributions - Task I and Task II

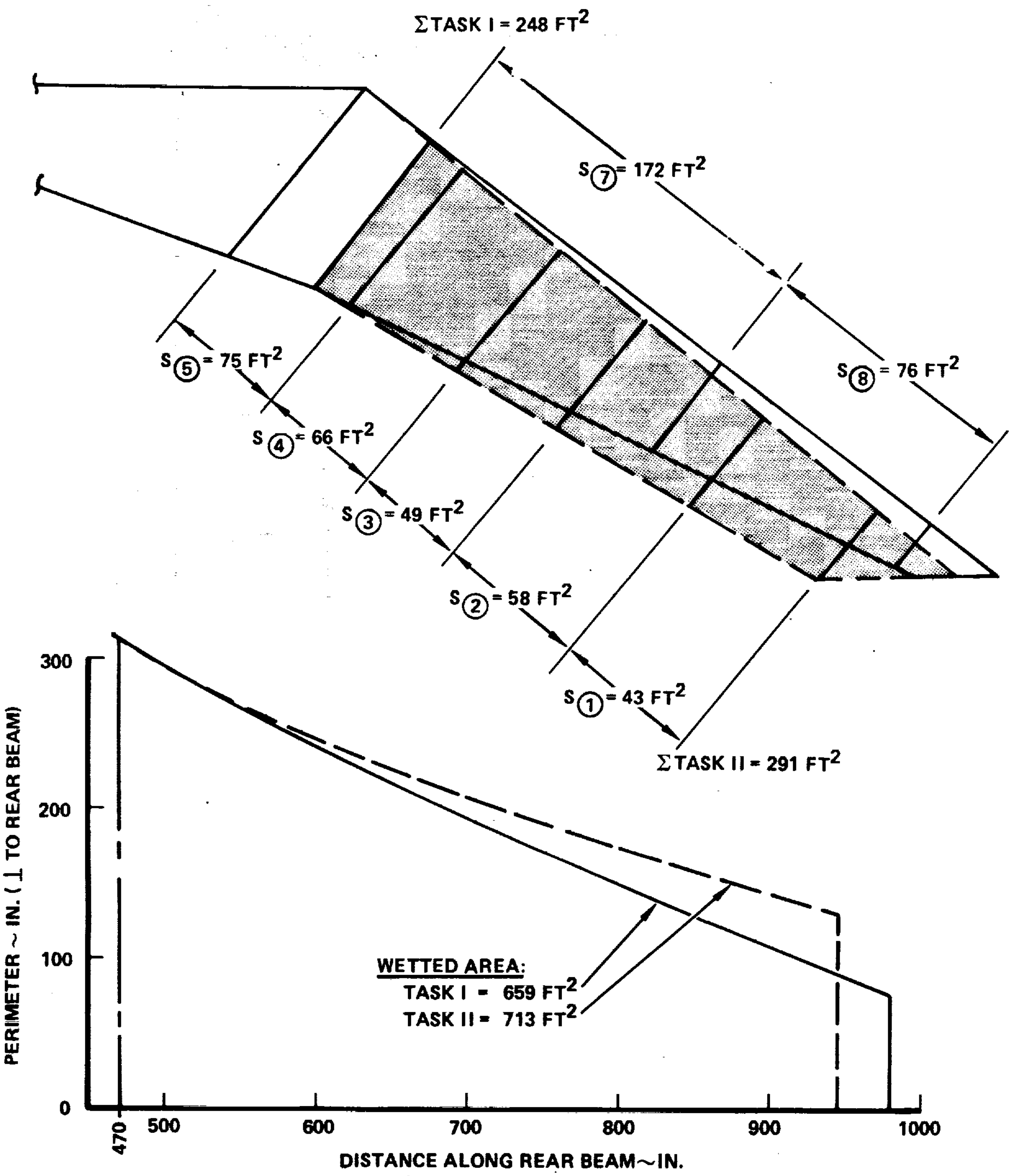
ΣTASK I = 248 FT^2
S⑦ = 172 FT^2
S⑧ = 76 FT^2
S⑤ = 75 FT^2
S④ = 66 FT^2
S③ = 49 FT^2
S② = 58 FT^2
S① = 43 FT^2
ΣTASK II = 291 FT^2
PERIMETER ~ IN. (⊥ TO REAR BEAM)
300
200
100
0
WETTED AREA:
TASK I = 659 FT^2
TASK II = 713 FT^2
470
500
600
700
800
900
1000
DISTANCE ALONG REAR BEAM~IN.
(TO MIDPOINT OF TIP)

Figure 15-28. Wing Tip Box Geometry - Task I and Task II

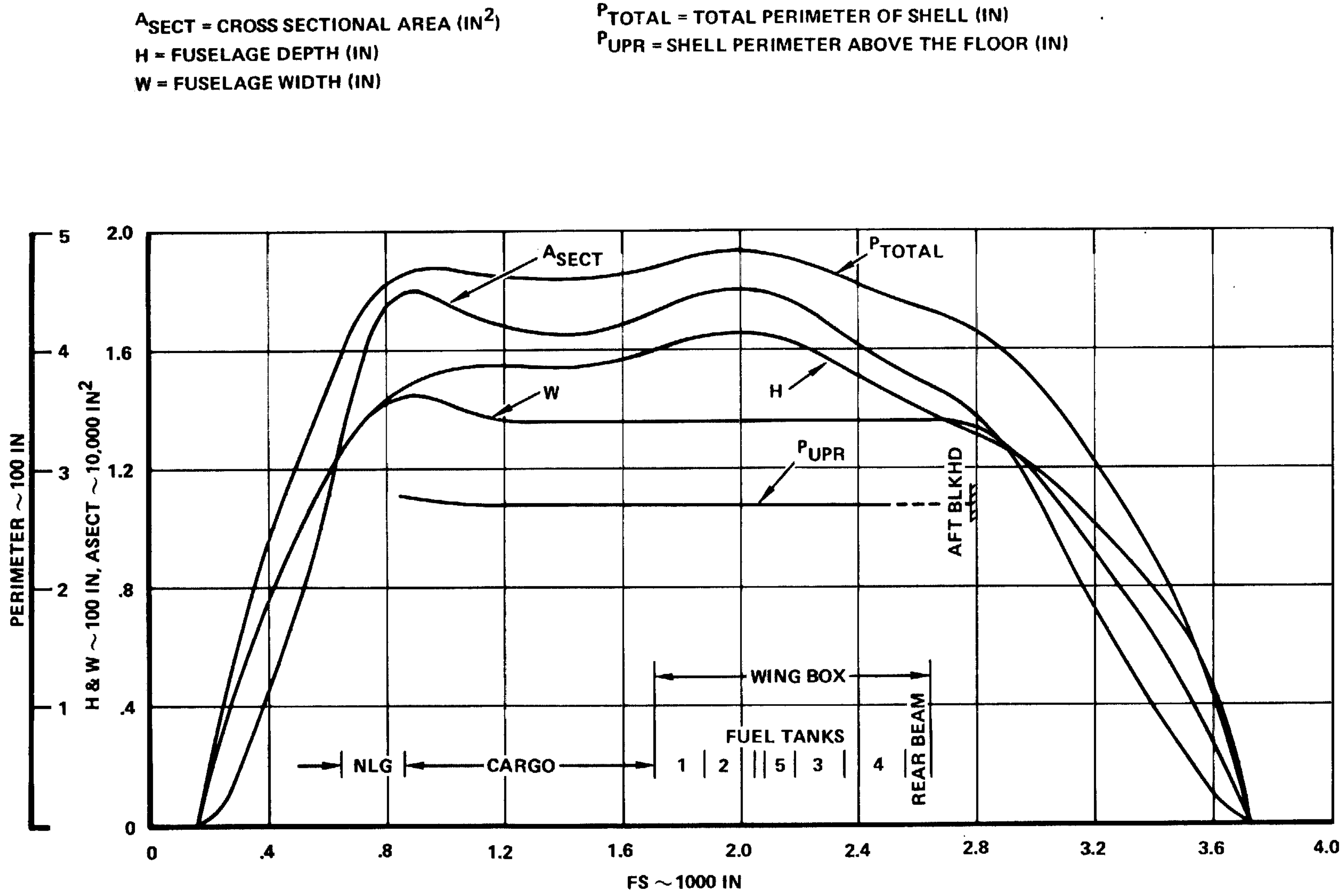

Figure 15-29. Body Geometry - Task I

segments. In the region of the wing box carry-through structure the body area is the circular arc segment above the upper wing surface. The segment areas and skin and stiffener unit weights are illustrated in Figure 15-30. The zee, open-hat and closed-hat stiffened concepts were analyzed at Fuselage Stations 2000, 2500 and 3000. As indicated on the figure, the closed-hat is 5- to 10-percent lighter and is selected for the weight calculation. The forward cabin utilizes zee stiffening and the transition assumes straight line extrapolation between the analysis points.

FUSELAGE STRUCTURE DETAILED ANALYSIS

For the detailed concept analysis, complete frame data and revised data for the closed-hat panel design were determined. The results of the analysis are identified on Table 15-40 in the shaded-column. The total fuselage weight for the zee- and open-hat stiffened shell are also shown. These data reflect the initial screening panel weights combined with the newly determined frame data for the closed-hat design. The weight trends, as indicated for the initial screening data for the panel concepts in Figure 15-30, are 5- to 6-percent lighter for the selected design concept. A minimum skin gage of .050-inch was used for the aft fuselage to meet the preliminary sonic fatigue requirements. Figure 15-31 and Table 15-41 illustrate the method used to derive the detailed fuselage shell weight for the closed-hat design. The mass data includes a non-optimum factor (NOF) applied to the shell structure unit weight to arrive at a typical estimate of the "as-constructed" weight. These non-optimum allowances are itemized as:

	Non-Optimum Factor (NOF)
Joints and splices	4-percent
Margins of safety (average)	3-percent
Sheet tolerances	2-percent
Access provisions	2-percent
Finish, sealant, misc.	3-percent
Total NOF	14-percent

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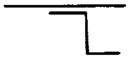
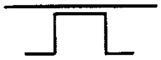
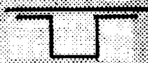
Use of these allowances, as described earlier in the Wing Structure Mass-Initial Screening section, means that a stress analysis which indicates a 3-pounds-per-square-foot panel yields an estimated fabricated weight, (1.14 x 3.0) = 3.52 pounds per square foot.

TABLE 15-40. FUSELAGE CONCEPT WEIGHTS - DETAILED CONCEPT ANALYSIS

AVERAGE SHELL MASS = $0.232 (w_{750} + w_{2000} + w_{2500} + w_{3000} + w_{3723})$

				
OPTIMUM UNIT MASS (psf) AT STATION:	750	1.56	1.56	1.54
	2000	3.54	3.51	3.27
	2500	4.03	3.86	3.53
	3000	3.54	3.51	3.43
0.05 inch (.064 mm) FOR SONIC FATIGUE	3723	2.15	2.15	2.15
AVERAGE SHELL MASS (INCL. NOF), W_{SHELL}	(lb/ft^2) (kg/m^2)	3.44 (16.80)	3.39 (16.55)	3.23 (15.77)
SHELL AREA (FS 690 TO 3723) = 7,167 ft^2 (666 m^2)				
SHELL MASS =	(lb) (kg)	24,654 (11,183)	24,296 (11,020)	23,148 (10,500)
FIXED MASS				
NOSE AND FLIGHT STATION		2,500	19,540 (8,863)	19,540 (8,863)
NLG WELL		900		
WINDSHIELD AND WINDOWS		1,680		
FLOORING AND SUPTS.		3,820		
DOORS AND MECHANISM		4,170		
UNDERWING FAIRING		1,870		
CARGO COMPARTMENT PROV.		1,060		
WING/BODY FITTINGS		1,500		
TAIL/BODY FITTINGS		600		
PROV. FOR SYSTEMS		740		
FINISH AND SEALING		700		
TOTAL FUSELAGE MASS (lb) (kg)		44,194 (20,046)	43,836 (19,884)	42,688 (19,[illegible])

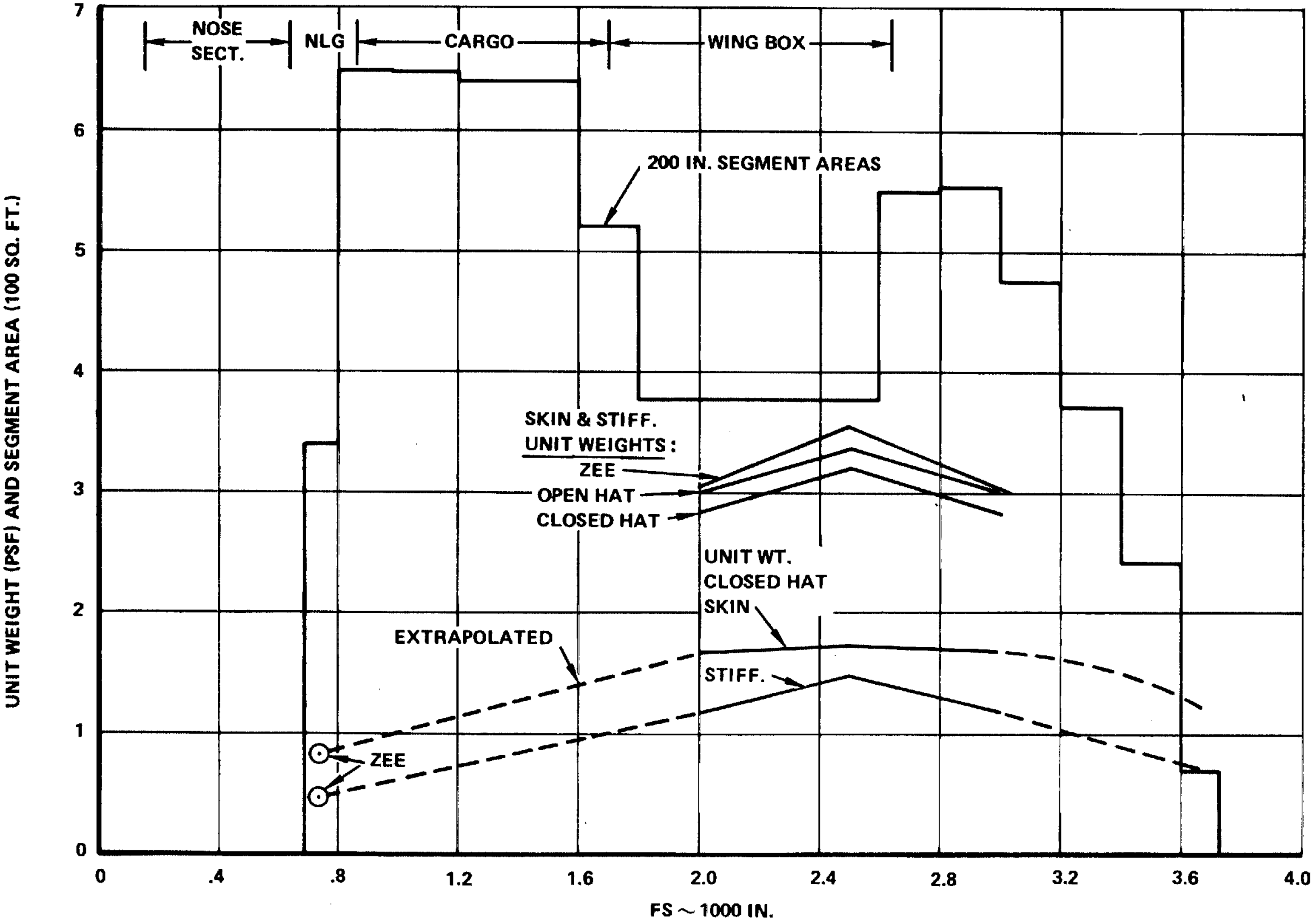


Figure 15-30. Skin and Stiffener Unit Weights - Initial Screening

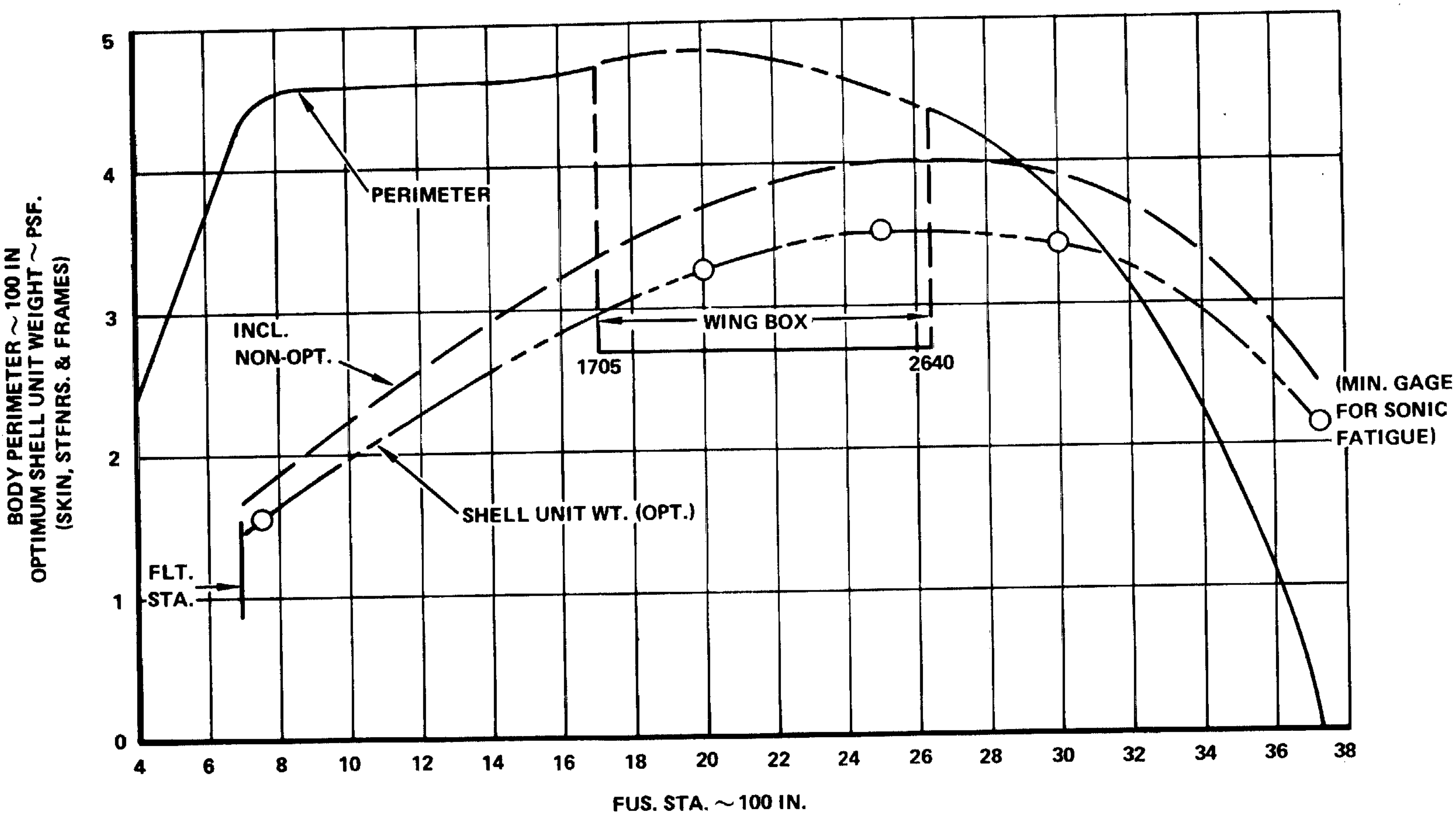


Figure 15-31. Unit Weight for Closed Hat Stringer Design - Detailed Concept Analysis

TABLE 15-41. FUSELAGE SHELL WEIGHTS - CLOSED HAT-STIFFENED PANEL CONCEPT - TASK I

SECT. F.S.	PERIMETER(A) (in.)	ΔS (ft^2)	SHELL UNIT WT. (psf)	ΔW_{OPT} (lb)	C.G. (in.)
690 - 800	445	340	1.55	527	745
800 - 1000	457	635	1.80	1143	900
- 1200	457	635	2.12	1346	1100
- 1400	460	639	2.42	1546	1300
- 1600	462	642	2.70	1733	1500
- 1705	467	340	2.90	986	1653
1705 - 2640	270	1753	3.35	5873	2175
2640 - 2800	425	472	3.50	1652	2720
- 3000	397	551	3.45	1901	2900
- 3200	347	482	3.35	1615	3100
- 3400	272	378	3.13	1183	3300
- 3600	173	240	2.75	660	3500
- 3723	70	60	2.33	140	3660
TOTALS		7167(B)		20,305(C)	2150

(A) AVERAGE DATA FOR SHELL

(B) CUTOUT FOR WING EXCLUDED

(C) OPTIMUM WEIGHT; "AS-FABRICATED" WEIGHT = 1.14 X 20,305 = 23,148 lb (SKINS = 11,423 lb; STIFFENERS = 8,120 lb; FRAMES = 3,605 lb)

ASSET PROGRAM RESULTS - TASK I

The ASSET Program synthesizes the vehicle so that it may be scaled up and down in size. The selected wing concepts are reflected by input coefficients to the wing group equation. The coefficients also consider those lightly loaded areas which are minimum gage, in addition to the primary load carrying structure.

Wing and fuselage weights resulting from the detailed concept analysis effort of Task I are reflected in the ASSET Program printouts. Data are presented for: (1) the aircraft scaled up to meet the range objective of 4200 nautical miles (7778 kilometers) and (2) the aircraft which has a takeoff weight of 750,000 pounds (340,000 kilograms) with variable range as dictated by the available fuel.

Table 15-42 summarizes the results of the ASSET runs for easy reference.

The wing concepts are described below:

1. Chordwise stiffened, Ti-6Al-4V Convex-beaded - mechanical fasteners
2. Spanwise stiffened, Ti-6Al-4V Hat - mechanical fasteners
3. Monocoque, Aluminum brazed honeycomb sandwich - mechanical fasteners
4. Monocoque, Aluminum brazed honeycomb sandwich - welded
5. Chordwise Stiffened, Convex-beaded, B/PI, composite reinforced spars
6. Best Combination:
    - Concept 5 for forward and aft box structure
    - Concept 3 for outer box structure

In all cases, the wing loading at takeoff (W_{TO}/S) is 69.3 pounds per square foot and the thrust loading (T/W_{TO}) is 0.477 pounds of thrust per pound of takeoff weight. This is an uninstalled S.L.S. thrust of 89,466 pounds per engine, increasing engine, nacelle and air induction weights over those carried for Task I.

FINAL DESIGN AIRPLANE MASS ESTIMATES

Detailed weight descriptions of the wing and fuselage for the Final Design airplane are given in Tables 15-43 and 15-44, respectively.

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TABLE 15-42. AIRCRAFT WEIGHT AND COST SUMMARY

WING CONCEPT	1		2		3		4		5		6	
RANGE, n.mi.	4,200	3,830	4,200	3,870	4,200	4,123	4,200	4,066	4,200	4,166	4,200	4,183
WEIGHTS (lb)												
TAKEOFF	884,847	750,000	867,126	750,000	772,641	750,000	789,992	750,000	759,498	750,000	754,665	750,000
FUEL AVAIL.	454,156	371,525	444,772	372,685	399,532	385,018	407,654	382,241	394,017	387,875	391,702	388,678
ZERO FUEL	430,691	378,475	422,353	377,315	373,109	364,982	382,338	367,759	365,481	362,125	362,963	361,322
PAYLOAD	49,000											49,000
OPERATING	381,691	329,475	373,353	328,315	324,109	315,982	333,338	318,759	316,481	313,125	313,963	312,322
OP. AND STD. ITEMS	11,448	10,653	11,357	10,564	10,915	10,776	10,995	10,751	10,861	10,802	10,839	10,809
EMPTY	370,243	318,821	361,997	317,651	313,194	305,206	322,343	308,008	305,620	302,323	303,125	301,513
WING	129,895	106,010	125,254	104,834	95,682	92,330	101,296	95,146	90,785	89,434	89,276	88,620
TAIL	14,404	11,339	13,986	11,339	11,834	11,339	12,220	11,339	11,545	11,339	11,440	11,339
BODY	46,109	42,688	45,675	42,688	43,283	42,688	43,733	42,688	42,939	42,688	42,812	42,688
LANDING GEAR	34,700	30,401	34,143	30,401	31,133	30,401	31,691	30,401	30,708	30,401	30,552	30,401
SURF. CONTR.	8,726	8,500	8,697	8,500	8,539	8,500	8,569	8,500	8,517	8,500	8,508	8,500
NACELLES	6,626	5,616	6,494	5,616	5,786	5,616	5,916	5,616	5,688	5,616	5,651	5,616
PROPULSION												
ENGINES	60,316	51,124	59,108	51,124	52,667	51,124	53,850	51,124	51,772	51,124	51,442	51,124
AIR INDUCT.	24,977	20,755	24,418	20,755	21,458	20,755	21,998	20,755	21,050	20,755	20,900	20,755
FUEL SYST.	6,326	5,598	6,239	5,603	5,789	5,662	5,872	5,649	5,729	5,676	5,706	5,679
ENG. CONTR.	1,744	1,631	1,729	1,631	1,650	1,631	1,664	1,631	1,639	1,631	1,635	1,631
INSTRUMENTS	1,365	1,230	1,347	1,230	1,253	1,230	1,270	1,230	1,239	1,230	1,235	1,230
HYDRAULIC	6,725	5,700	6,590	5,700	5,872	5,700	6,004	5,700	5,772	5,700	5,735	5,700
ELECTRICAL	4,650	4,550	4,637	4,550	4,567	4,550	4,581	4,550	4,557	4,550	4,554	4,550
AVIONICS	1,900											1,900
FURNISHINGS	11,500											11,500
E.C.S.	8,300											8,300
AUX. GEAR	1,980											1,980
A.M.P.R.	291,099	250,440	284,264	249,269	243,002	236,824	250,764	239,627	236,479	233,942	234,370	233,131
FLYAWAY COST (MIL. $)	103.2	90.6	99.8	89.2	107.5	104.9	109.4	104.8	94.7	93.8	94.0	93.6

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TABLE 15-43. FINAL DESIGN AIRPLANE - WING MASS ESTIMATE

ITEM		WEIGHT (lb)
VARIABLE WEIGHT		49,232(A)(B)
FORWARD BOX (PLANFORM AREA = 4136.6 ft^2)		(20,580)
• SURFACES ∼ CONVEX BEADED, CHORDWISE STIFFENED		9,452
• SPARS ∼ INCLUDING 522 lb COMPOSITES		8,558
• RIBS		2,570
AFT BOX (PLANFORM AREA = 2132.4 ft^2)		(17,384)
• SURFACES ∼ CONVEX BEADED, CHORDWISE STIFFENED		7,302
• SPARS ∼ INCLUDING 3,762 lb COMPOSITES		8,568
• RIBS		1,514
TRANSITION ∼ AFT BOX TO TIP BOX		(1,380)
TIP BOX (PLANFORM AREA = 947 ft^2)		(9,888)
• SURFACES ∼ BRAZED HONEYCOMB SAND., MECH. FAST.		8,235
• SPARS		1,336
• RIBS		317
FIXED WEIGHT	PLANFORM AREA (ft^2)	41,352
LEADING EDGE	1,047	5,235
TRAILING EDGE	1,941	4,888
WING/BODY FAIRING	800	1,600
LEADING EDGE FLAPS/SLATS	133	1,130
TRAILING EDGE FLAPS/FLAPERONS	553	5,890
AILERONS	250	1,250
SPOILERS	225	1,360
MAIN LANDING GEAR ∼ DOORS	484	2,904
SUP'T. STRUCTURE		3,750
B.L. 62 RIBS		1,430
B.L. 470 RIBS		700
FIN ATTACH RIBS (B.L. 602)		435
REAR SPAR		3,400
ENGINE SUPPORT STRUCTURE		3,580
FUEL BULKHEADS		3,800
TOTAL WING WEIGHT		90,584

(A) INCLUDES FAIL-SAFE PENALTY OF 822 lb

(B) INCLUDES COMPOSITE MATERIAL WEIGHT OF 4,284 lb

TABLE 15-44. FINAL DESIGN AIRPLANE - FUSELAGE MASS ESTIMATE

ITEM	WEIGHT (lb)
SHELL STRUCTURE	22,582$^{(A)}$
SKIN	11,144
STIFFENERS	7,921
FRAMES	3,517
FIXED WEIGHT	19,540
NOSE AND FLIGHT STATION	2,500
NOSE LANDING GEAR WELL	900
WINDSHIELD AND WINDOWS	1,680
FLOORING AND SUPPORTS	3,820
DOORS AND MECHANISM	4,170
UNDERWING FAIRING	1,870
CARGO COMPARTMENT PROV.	1,060
WING TO BODY FRAMES AND FITTINGS	1,500
TAIL TO BODY FRAMES AND FITTINGS	600
PROV. FOR SYSTEMS	740
FINISH AND SEALANT	700
TOTAL FUSELAGE WEIGHT	42,122

(A) INCLUDES FAIL-SAFE PENALTY OF 1,432 lb

The wing box (variable weight) represents the best combination of structural concepts, that is:

- Forward and aft box: convex beaded, chordwise stiffened surfaces with composite reinforced spar caps.
- Outer box: brazed honeycomb sandwich surfaces, mechanically fastened closures.

The weight description includes fail-safe provisions, allowances for flutter prevention, and panel thickness changes for manufacturing/design constraints. The fixed weight consists of those items which are unaffected by box structural concept, such as surface controls, engine rails, leading and trailing edge structure. Items in this category are weighed by comparison with previous supersonic cruise aircraft and contemporary aircraft.

The fuselage weight is also divided into two major categories, shell weight and fixed weight. Here again the shell weight is dependent upon structural concept, while the fixed weight such as doors, windows, flight station and fairing are unaffected. Fixed weight items are evaluated by comparison with L-2000 and contemporary aircraft.

WING STRUCTURE MASS

The wing structure mass for the Final Design Airplane is based on the NASTRAN strength-stiffness bulk data (Reference Section 9, Structural Analysis Models). Appropriate non-optimum factors are applied to the integrated mass data to obtain the "as-constructed" wing weights.

The effect of the various design parameters on the sizing of the structural elements are included in the determination of the bulk data flexibilities. The parameters include:

- Strength
- Fatigue Life
- Temperature
- Thermal Stress
- Flutter
- Aeroelastic Loads
- Jig-shape Effects on Loads
- Design & Manufacturing
- Material Selection (Including Composites)
- Minimum Gage

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The integrated mass for the wing surface panel, substructure and non-optimum factor are shown at each grid point of the wing planform of Figure 15-32. Mass summations for the forward, aft, and tip boxes and transition are shown. Appropriate allowance for fail-safe requirements at the transition of the monocoque tip design to the chordwise stiffened aft box are included.

Figure 15-33 presents a comparison of the forward and aft box weights for the strength and strength-stiffness design. The cumulative weights for the wing inboard of BL470 are presented. The data generated indicated that except for small deviation in the aft box region, the strength-design and the strength/stiffness design are essentially unchanged. The change in aft box weight is partially attributed to load increases from jig shape effects on aeroelastic loads. These aft box mass increases as well as the wing fixed weight items are reflected in the SIC distribution.

Table 15-45 presents the mass increments required to meet the specified fail-safe criteria. The unit weight data are as calculated in Section 13 and distributed to the aft box and wing tip structure according to the associated stress levels. The data indicates the aft box penalties appear to be localized in the region of the propulsion system installations. The wing tip penalties are associated with spanwise fail-safe straps required to stop propagation of a chordwise crack in both skins of the honeycomb sandwich panels. It appears that this penalty can be further minimized if the tip structure were specifically designed to meet the stiffness requirement to suppress flutter in light of the fail-safe design requirements. Application of boron-aluminum surface panels in lieu of titanium honeycomb sandwich and considering a multispar design to minimize chordwise crack growth offer significant potential mass reduction for the wing tip structure design.

Fuselage Structure Mass Estimates

The fuselage mass estimates for the Final Design airplane are based on the verification of the results of the Detailed Concept Analysis of Task I (refer to Figure 15-31). The "as-constructed" fuselage weights were established by application of a non-optimum factor to the shell (i.e., skin-stringer and frame) unit weight determined for the five selected fuselage stations. This data is again presented on Figure 15-34, as corrected for the shortened nose and slightly altered wing carry through structure. Superimposed on the figure are (1) the verified unit

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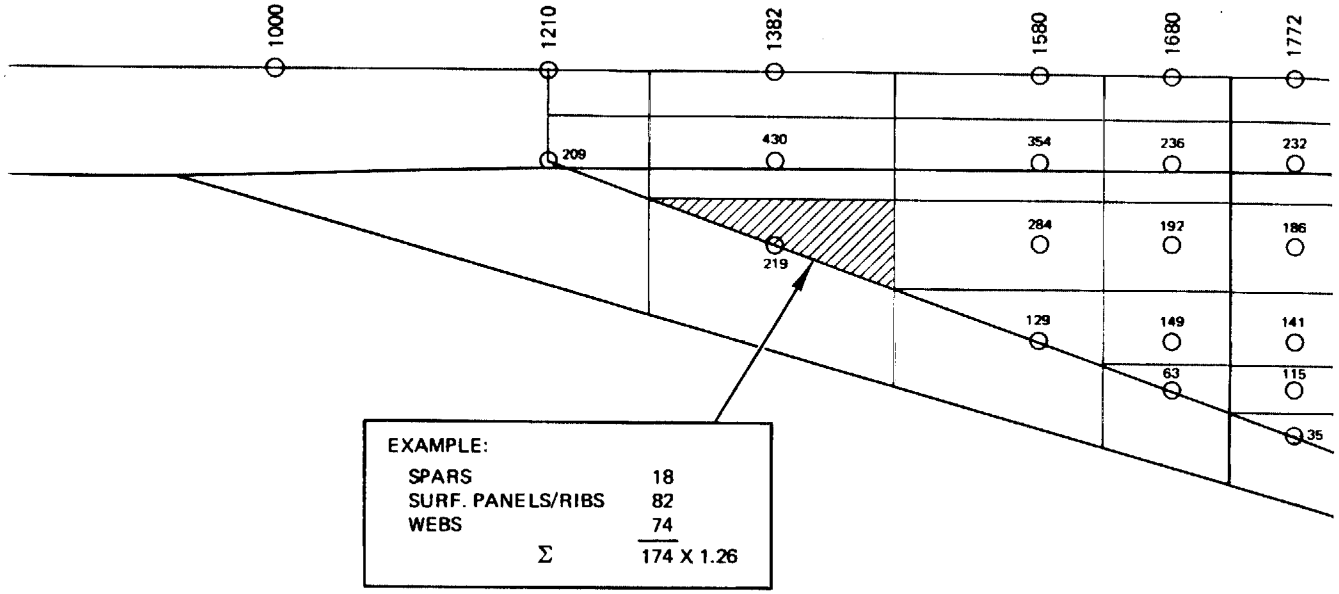
1000
1210
1382
1580
1680
1772
209
430
354
236
232
219
284
192
186
129
149
141
63
115
35

EXAMPLE:

SPARS	18
SURF. PANELS/RIBS	82
WEBS	74
Σ	174 X 1.26

$W_{FWD\ BOX}$ = 10,290 LB/SIDE

$W_{AFT\ BOX}$ = 8,692 LB/SIDE

$W_{TRANSITION}$ = 690 LB/SIDE (WEIGHTS IN PARENTHESIS)

$W_{TIP\ BOX}$ = 4944

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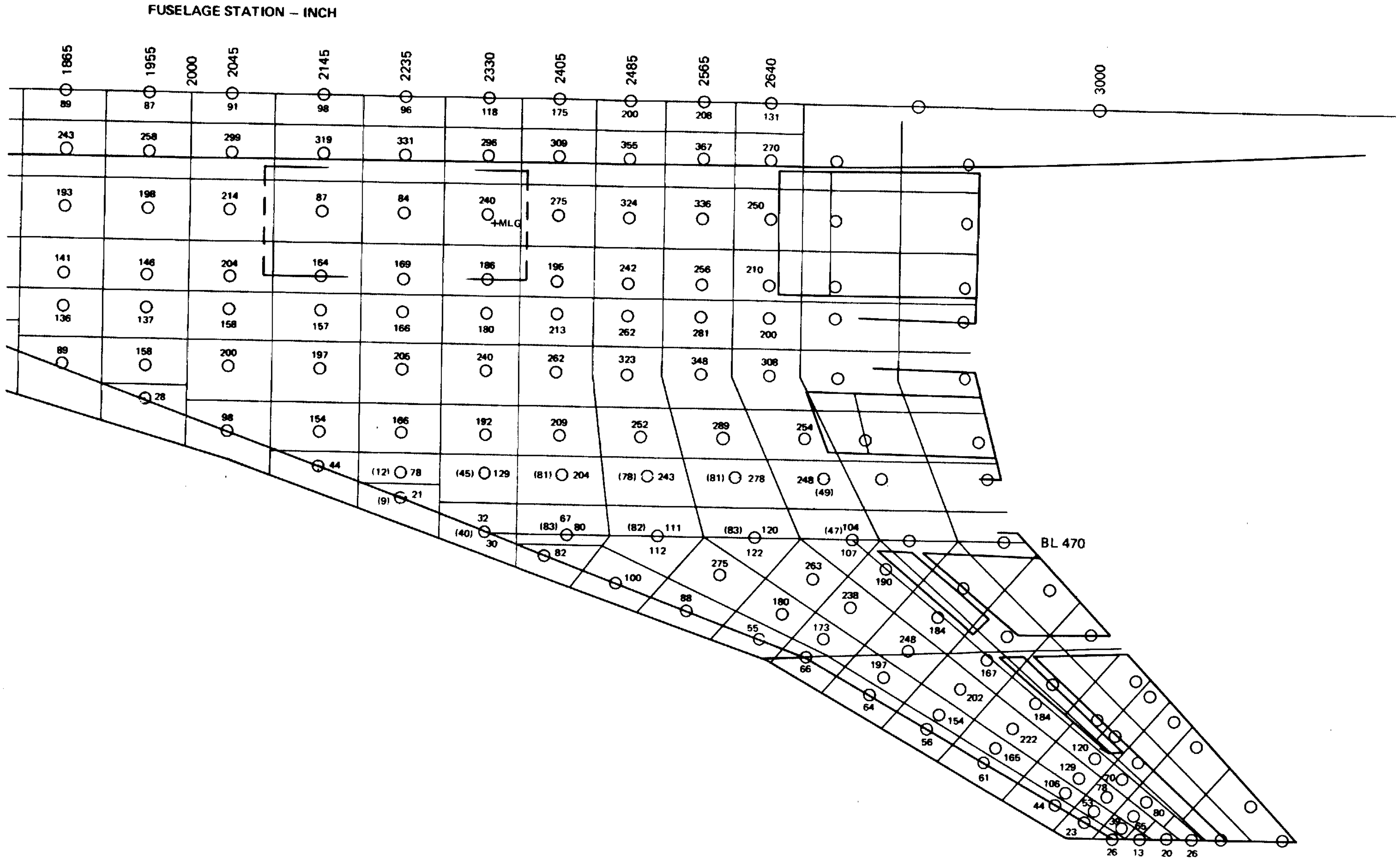
FUSELAGE STATION – INCH
1865
1955
2000
2045
2145
2235
2330
2405
2485
2565
2640
3000
89
87
91
98
96
118
175
200
208
131
243
258
299
319
331
296
309
355
367
270
193
198
214
87
84
240
275
324
336
250
+MLG
141
146
204
164
169
186
196
242
256
210
136
137
158
157
166
180
213
262
281
200
89
158
200
197
206
240
262
323
348
308
28
98
154
166
192
209
252
289
254
44
(12) 78
(45) 129
(81) 204
(78) 243
(81) 278
248 (49)
(9) 21
32
(40) 30
(83) 67 80
(82) 111
(83) 120
(47) 104
82
112
122
107
BL 470
100
275
263
190
88
180
238
184
55
173
248
66
197
167
64
202
184
154
56
222
120
165
129
61
106
70
78
53
80
44
39
65
23
26
13
20
26

Figure 15-32. F[illegible] Design Airplane - Wing Structure Mass At SIC Grid Point

FOLDOUT FRAME 3

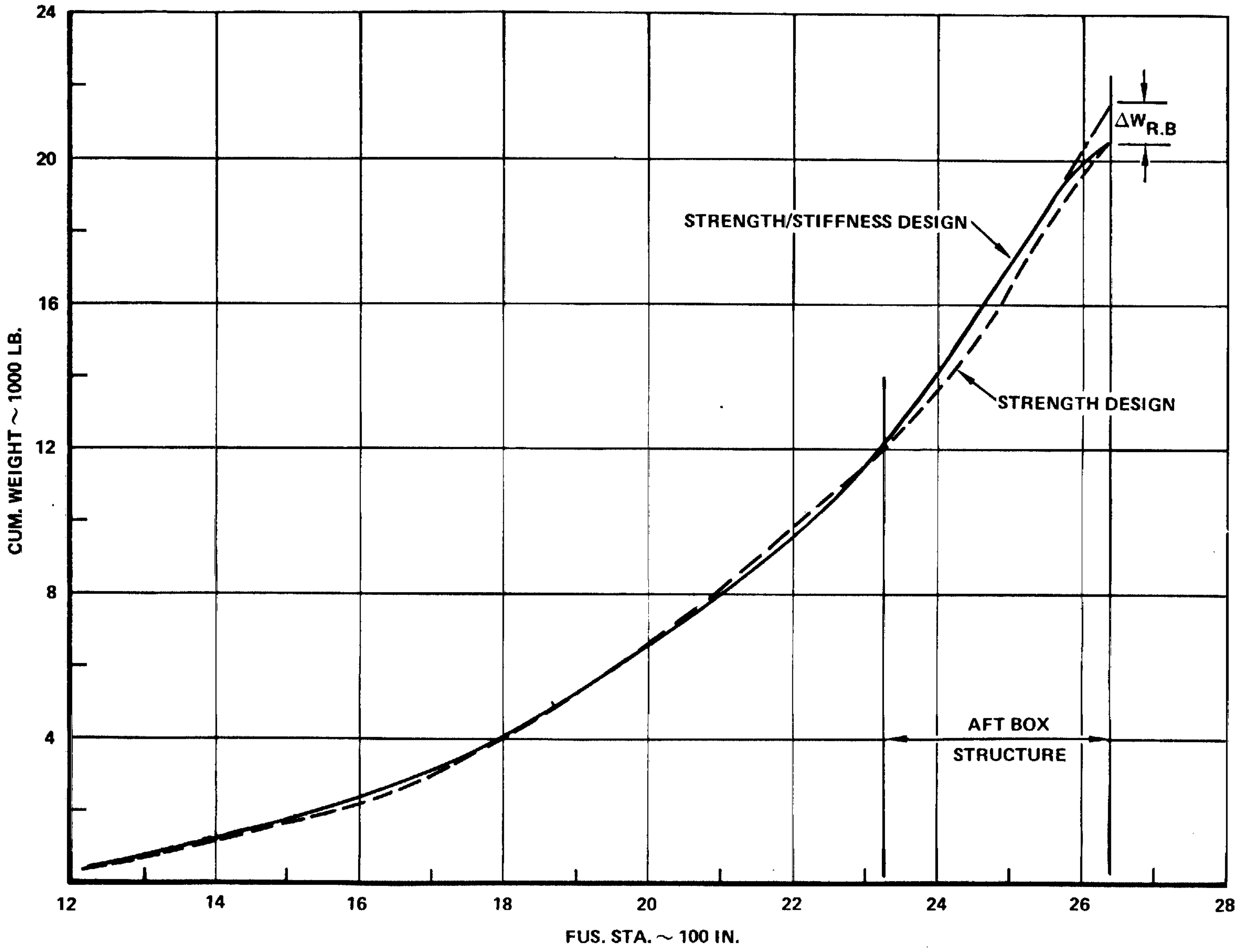

Figure 15-33. Comparison of Inboard Wing Mass - Task II

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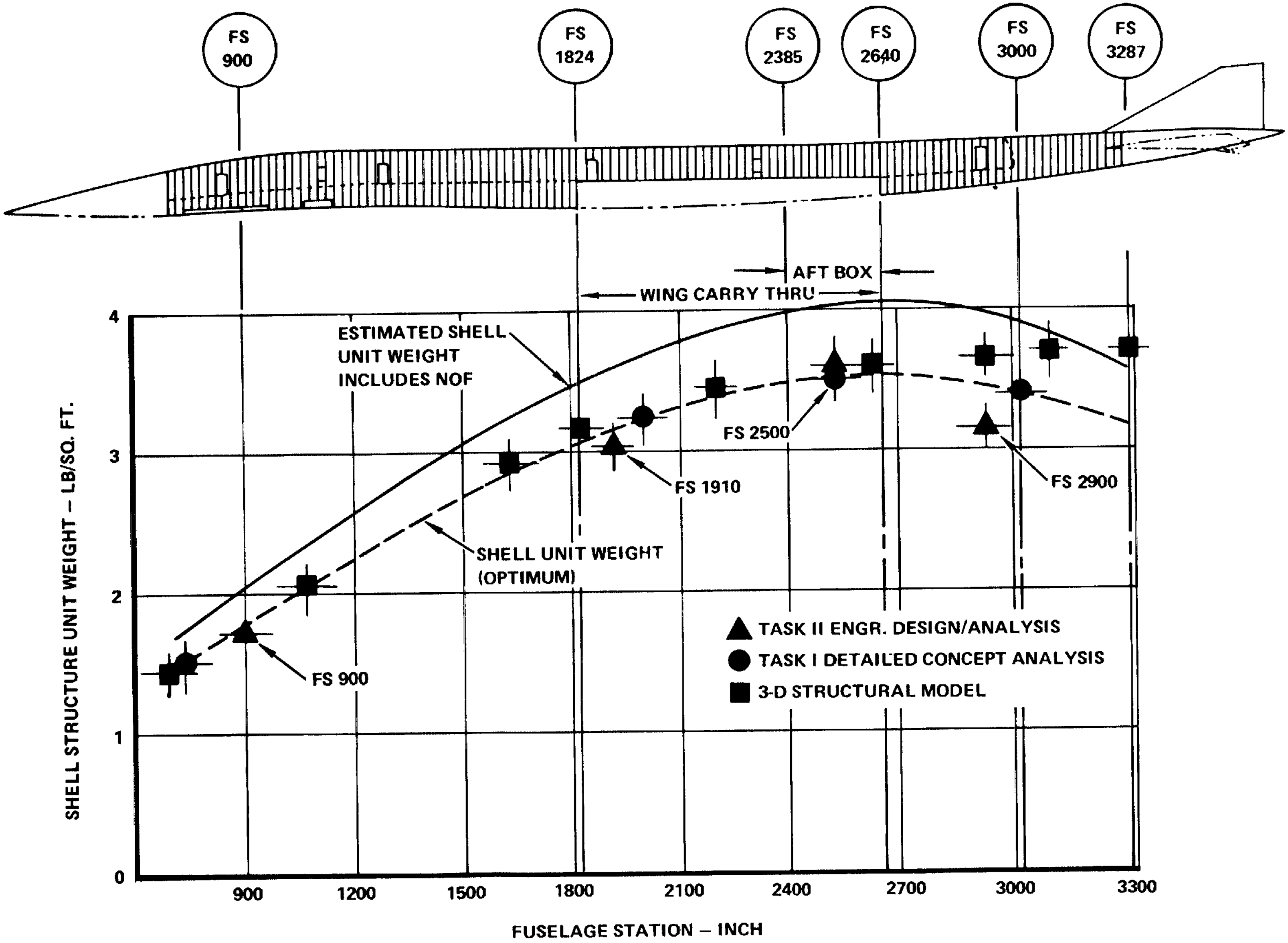

Figure 15-34. Final Design Airplane - Fuselage Shell Structure Mass

weights from the 3-D finite element structural model and (2) the unit weight calculations from detailed analysis of Task II which includes the strength and fail-safe requirements. The results of the detailed analysis (including strength and fail-safe design) are presented in Table 15-46 for F.S. 900, F.S. 1910, F.S. 2500 and F.S. 2900. The mass trends are very near the optimum shell weights calculated for Task I. In the aftbody, the results indicate values less than previously used. Rather than reduce the mass trends in the aftbody, the originally determined trends were maintained to provide adequate structural integrity for asymmetric conditions which were not fully investigated for the fuselage design.

Figure 15-35 presents the impact of composite reinforcement of the titanium hat section stringers with boron-aluminum and boron-polyimide. The results were constrained by frame and stringer spacing. Although approximately 5-percent reduction in shell structure unit weight is indicated on Figure 15-35, the effect on the total body weight is only slightly greater than 1-percent. The application of boron-polyimide shows slightly increased benefits over boron-aluminum. The Final Design airplane retains the basic skin-stringer-frame construction using titanium alloy 6Al-4V (annealed), although in the interior region (i.e., floors, floor beams, trim) epoxy resin composites are employed in selected areas.

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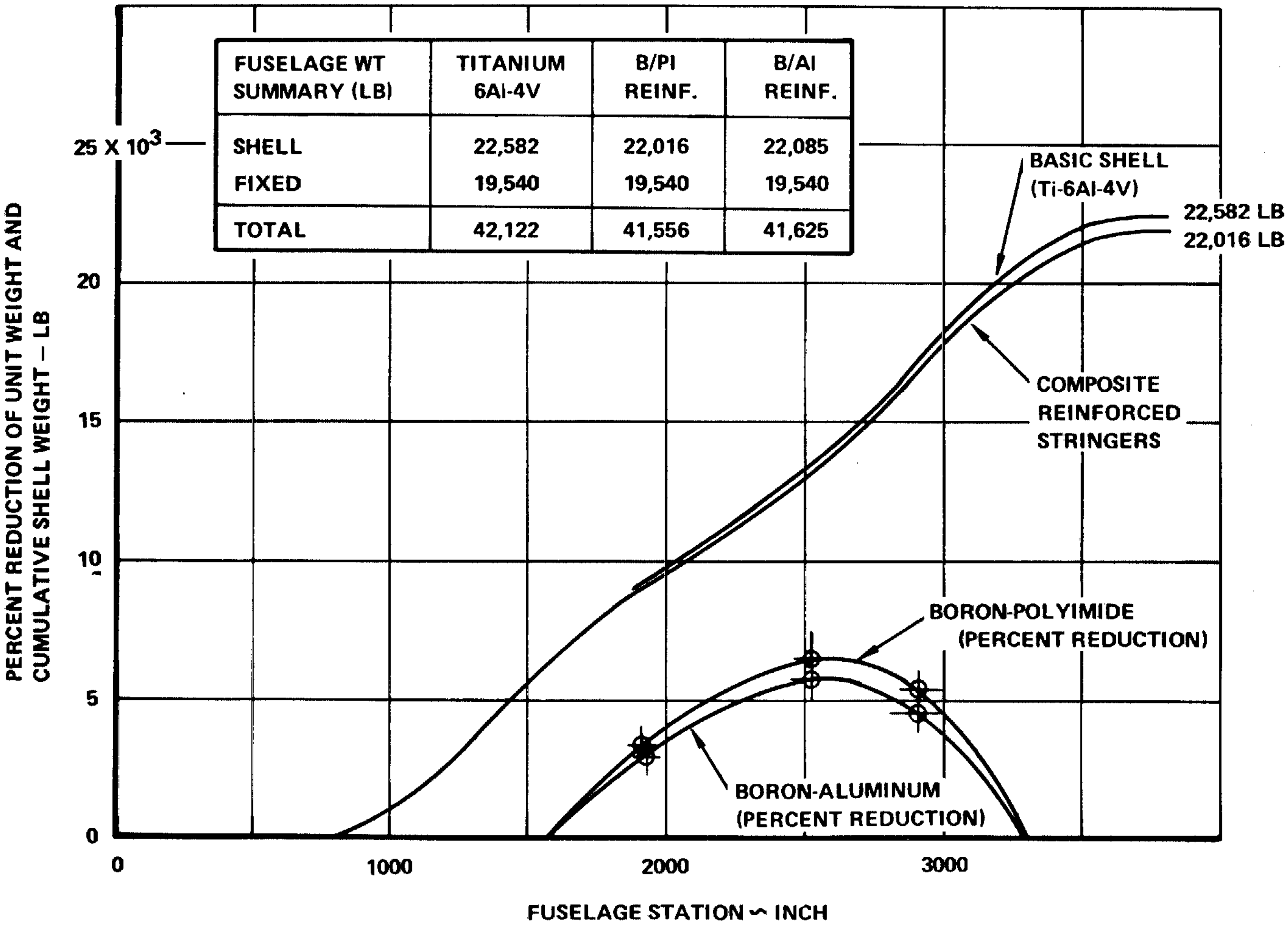
TABLE 15-45. WING STRUCTURE FAIL-SAFE PENALTY

WING STRUCTURE	POINT DESIGN	EFFECTIVE AREA (ft^2)	MASS INCREMENT	
			(lb/ft^2)	(lb)
AFT BOX	41036	264.4	1.39	368
	40536	209.0	1.63	340
	Σ			(708)
TIP BOX	41316	115.8	0.81	94
	41348	214.4	0.09	20
	Σ			(114)
TOTAL FAIL-SAFE PENALTY				(822)

TABLE 15-46. FINAL DESIGN AIRPLANE - FUSELAGE UNIT WEIGHTS

POINT DESIGN REGION	(1)FUSELAGE UNIT WEIGHTS (lb/ft^2)			
	PANEL	FRAME	FAIL-SAFE PENALTY	Σ TOTAL
FS 800 - 1000	1.29	0.22	0.25	1.76
FS 1865 - 1955	2.40	0.46	0.22	3.08
FS 2485 - 2565	2.53	0.51	0.46	3.50
FS 2800 - 3000	2.56	0.20	0.10	2.86

(1) EXPRESSED AS EQUIVALENT SURFACE PANEL WEIGHT



FUSELAGE WT SUMMARY (LB)	TITANIUM 6Al-4V	B/PI REINF.	B/Al REINF.
SHELL	22,582	22,016	22,085
FIXED	19,540	19,540	19,540
TOTAL	42,122	41,556	41,625

Figure 15-35. Impact of Composite Reinforcements on Fuselage Shell Structure Mass

REFERENCES

1. Mach 2.7 Fixed Wing SST Model 969-336C (SCAT-15F) D6A11666-1 and D6A11666-2, The Boeing Company, July 1969

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SECTION 16

PRODUCTION COSTS

BY

W. NEWCOMB

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SECTION 16

PRODUCTION COSTS

INTRODUCTION

The results of the analyses performed to develop production costs of the five wing structural arrangements are presented in this section.

The design concepts are identified as follows:

(1) Chordwise stiffened wing design (Mechanically fastened)

(2) Spanwise stiffened wing design (Mechanically fastened)

(3) Monocoque wing design (Mechanically fastened)

(4) Monocoque wing design (Jointed by welding)

(5) Composite reinforced wing design (Mechanically fastened)

Each of the wing designs were analyzed in sufficient depth to establish credible production costs estimates to be used for conducting simplified cost benefit trade studies for the evaluation and selection of the best structural approach for a March 2.7 arrow-wing supersonic cruise aircraft. These estimates included production manhours, material costs, and fabrication and assembly tool-make time. The production costs for each wing design were translated into "value per pound" inputs to the ASSET (Advanced Synthesis and Evaluation Technique) computer program to determine fly-away and total system costs. The ASSET computer program and its usage is described in Section 17, Concept Evaluation and Selection.

COST METHODOLOGY

To achieve useable cost inputs for each of the wing designs, it proved necessary to evaluate the fabrication and assembly operation for each of the details in the surface panels and substructure of various areas of the airframe as generalized in Figure 16-1. This type of analysis was essential since each wing design, having the same planform and basically the same material usage, differed in many ways.

These differences included part count, methods used to produce the surface panels and internal structure, plus design difference resulting from constraints imposed by the location on the wing planform.

Analyses to the required detail were achieved by conducting the costing effort from the component level to the major assembly level. Areas of the wing, coincident with the point design regions used for the structural analysis, were adopted as shown in Figure 16-2. Three regions were selected for cost analysis: 40322, 40536, and 41348. The sizing data (i.e. skin thickness, cap size, etc.) in these regions were considered as representative for the wing forward area, wing aft box area and the wing tip, respectively. The major assembly costs for the three areas of the wing were then used to estimate average costs for the total wing structure.

Production wing panel sizes were determined for each of the five wing designs and the panels surrounding the "point design regions" were selected for analyses. Figure 16-3 shows details associated with forward wing area of wing design (2). Note that the panel structure is divided into six elements: upper and lower skin assemblies and two different spar and rib designs. This structural breakdown is typical in each panel, with each wing area, (forward, aft, outer), for each wing design.

DETAIL COSTING

Cost analyses were performed for the variety of structural details shown on Figures 16-4 through 16-8. Production costs were estimated for (1) the weld bonded beaded panel design, (2) the weld bonded hat stiffened panel design and (3) the aluminum brazed honeycomb core sandwich design, using appropriate advanced producibility techniques (see Section 7, Materials and Producibility). The variety of spar and rib configuration costs were determined for each wing design considering such factors as metal removal and welding requirements. Production manhours were developed using the joint designs shown in the figures, consistent with each design.

Fabrication data for the upper and lower skin assemblies were estimated by the manhours and material weight per square foot of each panel. The fabrication data for the linear structure, such as caps, webs, etc., were determined by the lineal foot. All assembly data were based on type of joint design, such as number of fasteners, inches of weld, etc., and were also estimated by the lineal foot.

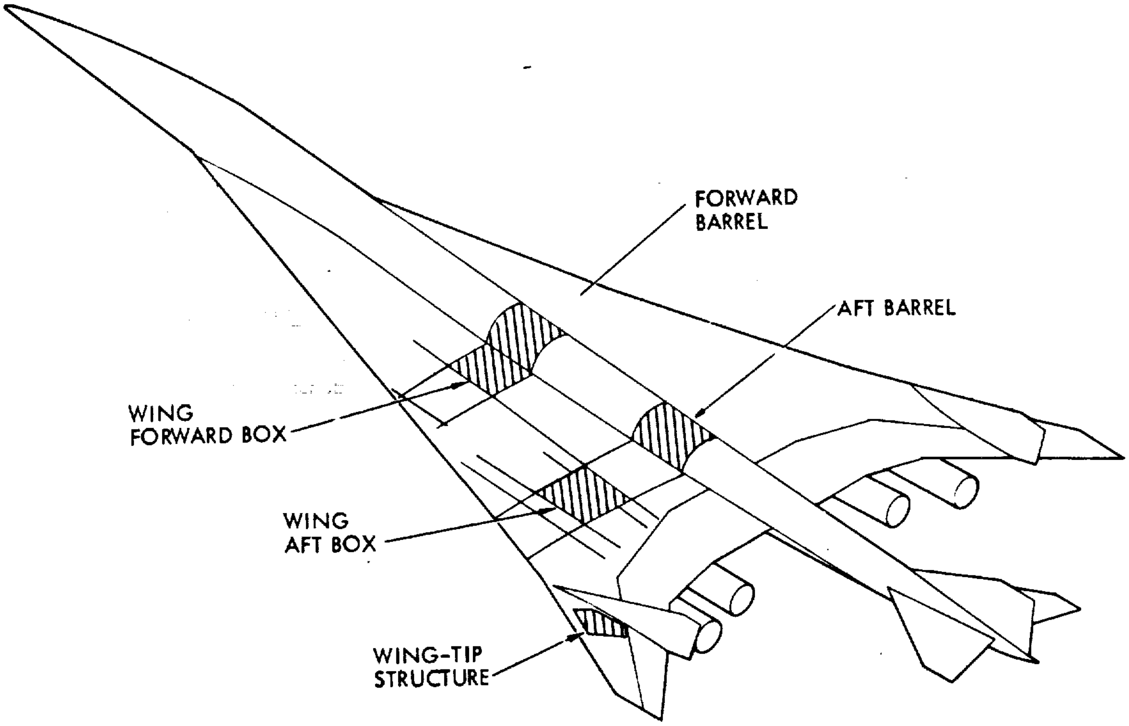

FORWARD BARREL
AFT BARREL
WING FORWARD BOX
WING AFT BOX
WING-TIP STRUCTURE

Figure 16-1. Cost Analysis Methodology

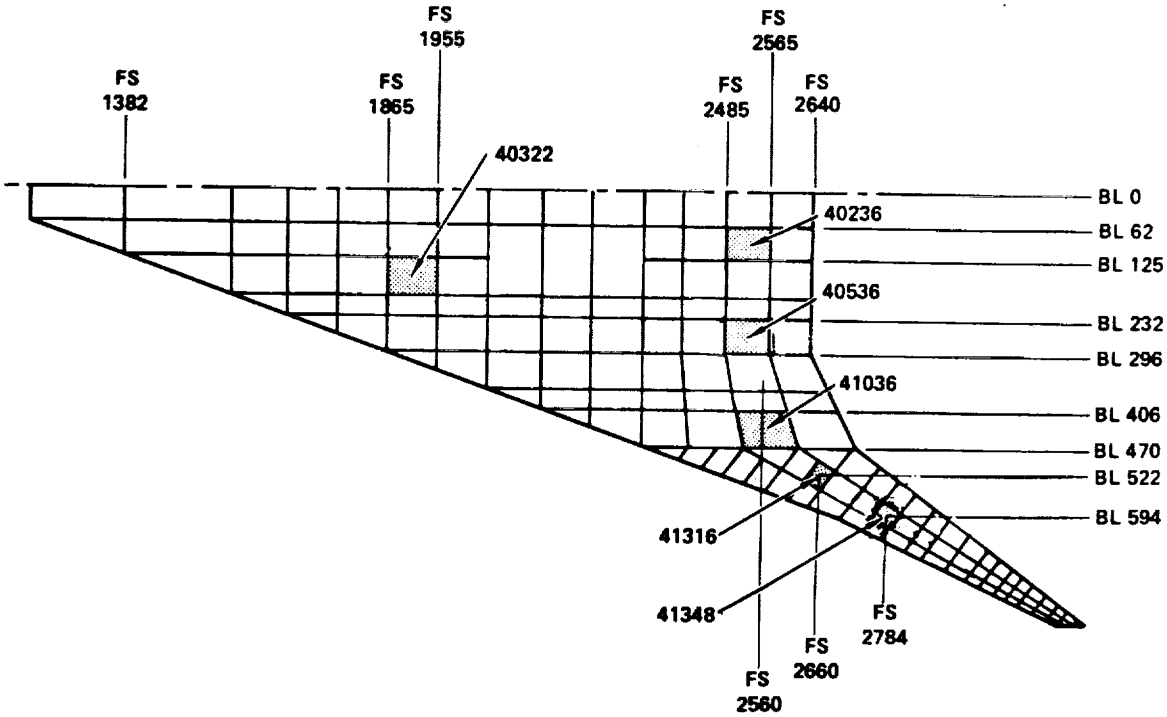
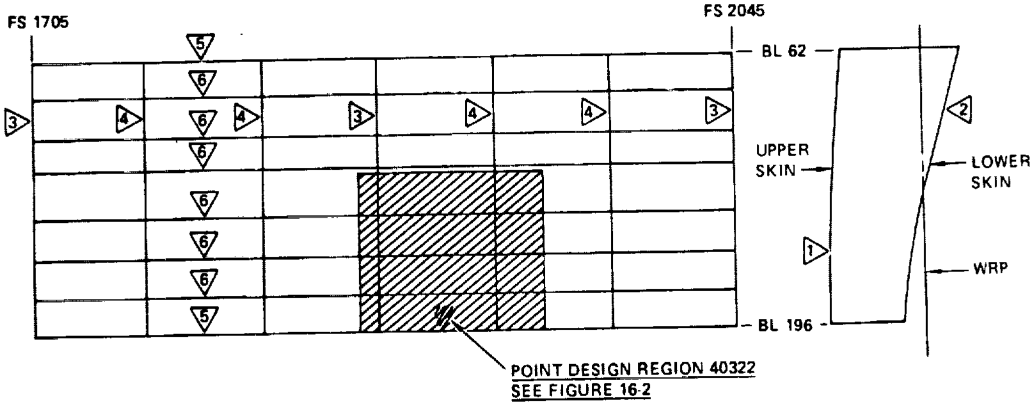

FS 1382
FS 1865
FS 1955
FS 2485
FS 2565
FS 2640
40322
40236
40536
41036
41316
41348
BL 0
BL 62
BL 125
BL 232
BL 296
BL 406
BL 470
BL 522
BL 594
FS 2784
FS 2660
FS 2560

Figure 16-2. Wing Point Design Regions


FS 1705
FS 2045
BL 62
BL 196
UPPER SKIN
LOWER SKIN
WRP
1
2
3
4
5
6
POINT DESIGN REGION 40322
SEE FIGURE 16-2

	ELEMENT	DIMENSIONAL DATA
1	UPPER SKIN	317 SQ. FT.
2	LOWER SKIN	317 SQ. FT.
3	CIRCULAR ARC SPARS	34 LIN. FT.
4	TRUSS SPARS	45 LIN. FT.
5	CIRCULAR ARC RIBS	57 LIN. FT.
6	TRUSS RIBS	170 LIN. FT.

Figure 16-3. Wing Forward Area - Cost Analysis Region

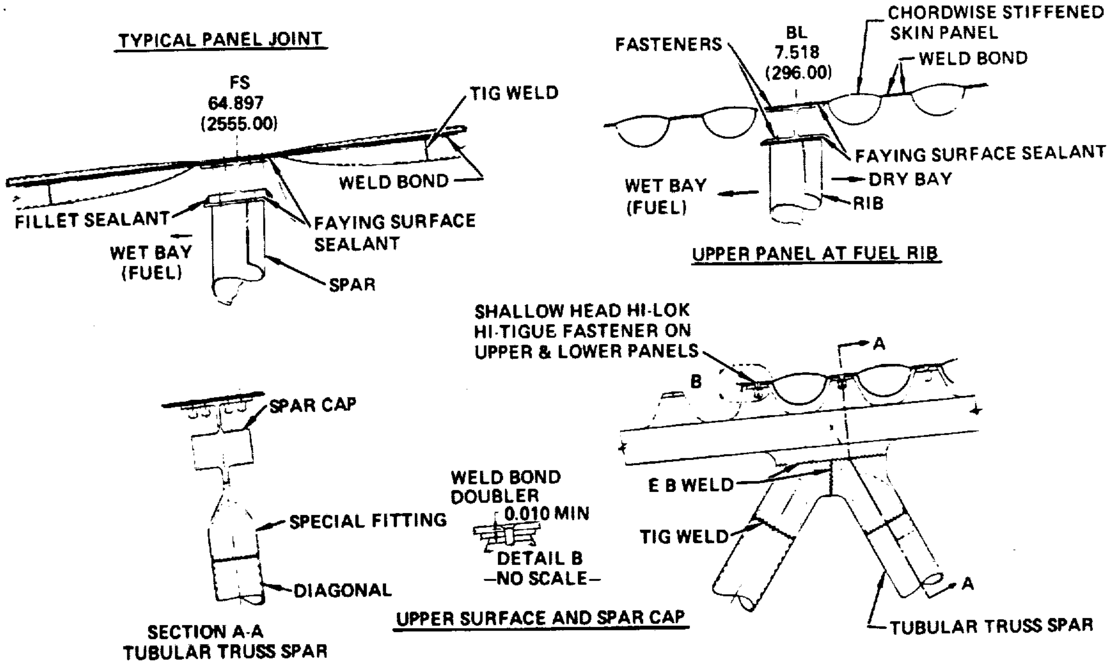

TYPICAL PANEL JOINT
FS
64.897
(2555.00)
TIG WELD
WELD BOND
FILLET SEALANT
FAYING SURFACE SEALANT
WET BAY (FUEL)
SPAR
FASTENERS
BL
7.518
(296.00)
CHORDWISE STIFFENED SKIN PANEL
WELD BOND
FAYING SURFACE SEALANT
WET BAY (FUEL)
DRY BAY
RIB
UPPER PANEL AT FUEL RIB
SHALLOW HEAD HI-LOK HI-TIGUE FASTENER ON UPPER & LOWER PANELS
A
B
SPAR CAP
SPECIAL FITTING
DIAGONAL
SECTION A-A
TUBULAR TRUSS SPAR
WELD BOND DOUBLER
0.010 MIN
DETAIL B
—NO SCALE—
UPPER SURFACE AND SPAR CAP
E B WELD
TIG WELD
A
TUBULAR TRUSS SPAR

Figure 16-4. Structural Details - Chordwise Stiffened Wing Design

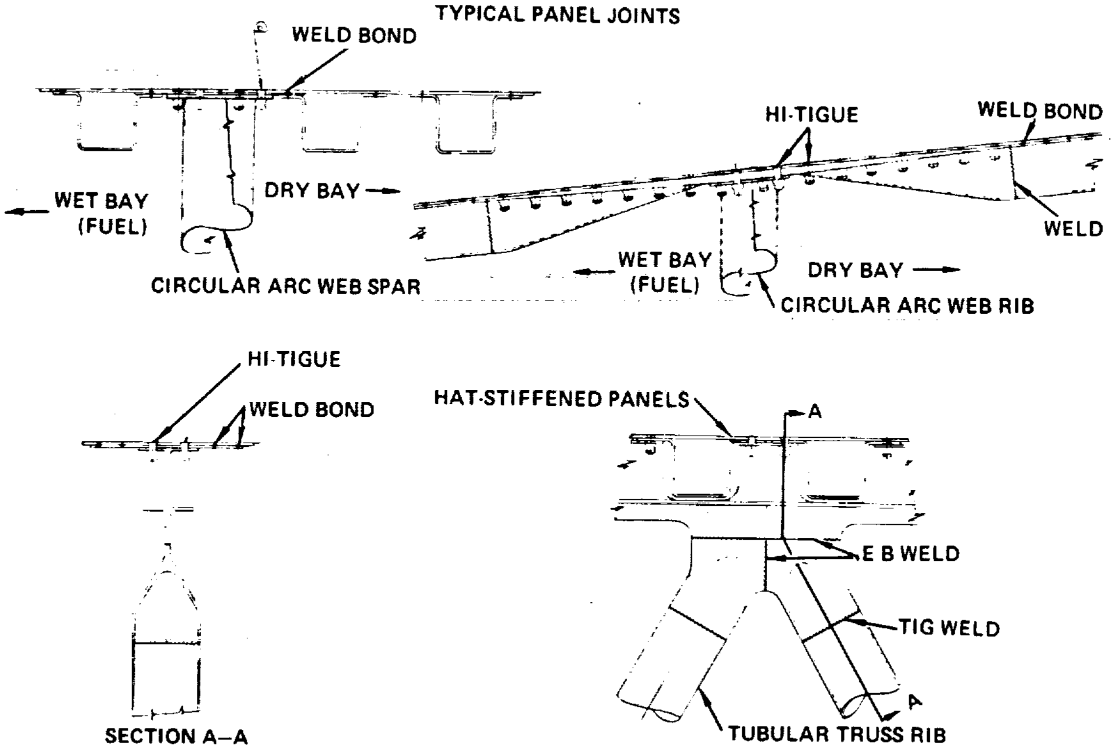

TYPICAL PANEL JOINTS
WELD BOND
WET BAY (FUEL)
DRY BAY
CIRCULAR ARC WEB SPAR
HI-TIGUE
WELD BOND
WELD
WET BAY (FUEL)
DRY BAY
CIRCULAR ARC WEB RIB
HI-TIGUE
WELD BOND
HAT-STIFFENED PANELS
A
E B WELD
TIG WELD
A
SECTION A–A
TUBULAR TRUSS RIB

Figure 16-5. Structural Details - Spanwise Stiffened Wing Design

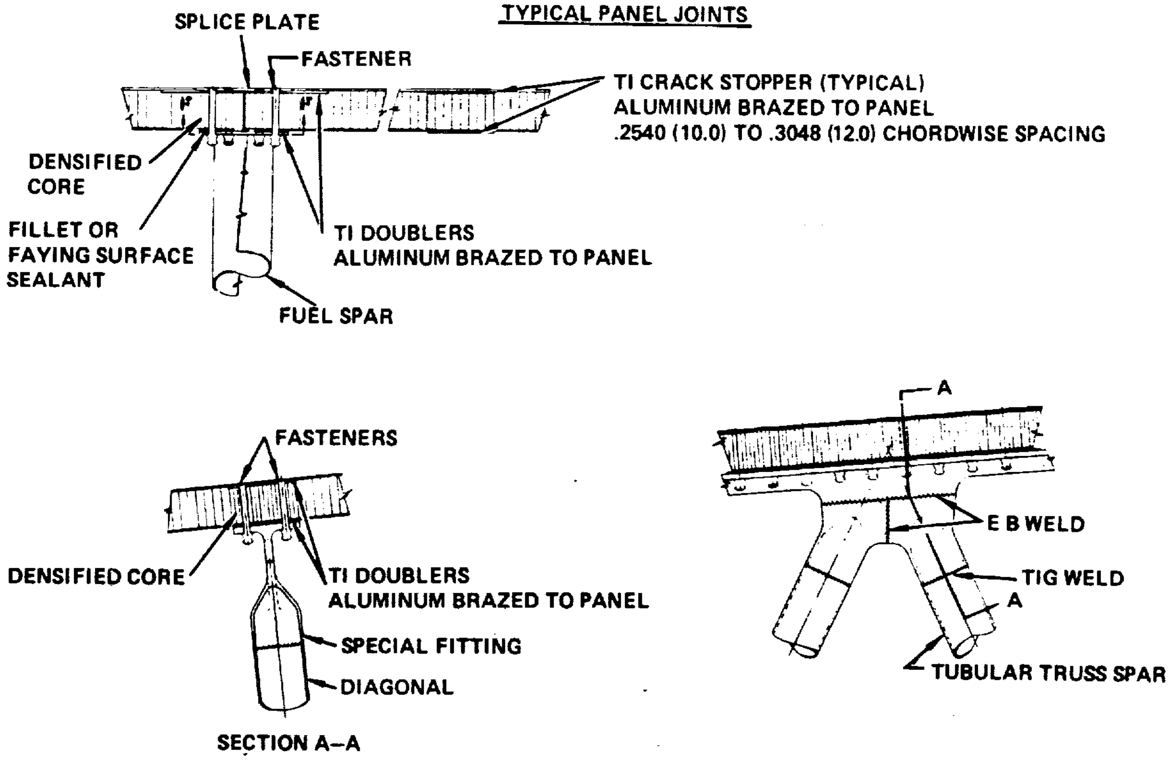

TYPICAL PANEL JOINTS
SPLICE PLATE
FASTENER
TI CRACK STOPPER (TYPICAL)
ALUMINUM BRAZED TO PANEL
.2540 (10.0) TO .3048 (12.0) CHORDWISE SPACING
DENSIFIED CORE
FILLET OR FAYING SURFACE SEALANT
TI DOUBLERS
ALUMINUM BRAZED TO PANEL
FUEL SPAR
FASTENERS
DENSIFIED CORE
TI DOUBLERS
ALUMINUM BRAZED TO PANEL
SPECIAL FITTING
DIAGONAL
SECTION A–A
A
E B WELD
TIG WELD
A
TUBULAR TRUSS SPAR

Figure 16-6. Structural Details - Monocoque Wing Design

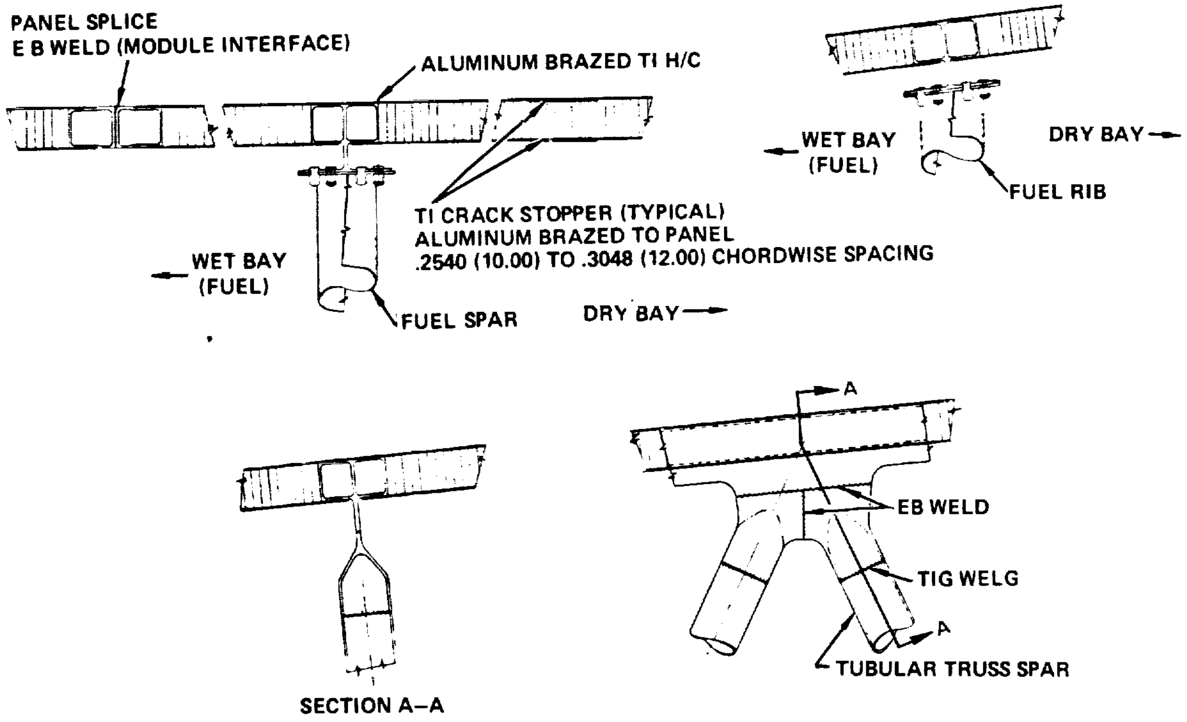

PANEL SPLICE
E B WELD (MODULE INTERFACE)
ALUMINUM BRAZED TI H/C
TI CRACK STOPPER (TYPICAL)
ALUMINUM BRAZED TO PANEL
.2540 (10.00) TO .3048 (12.00) CHORDWISE SPACING
WET BAY
(FUEL)
FUEL SPAR
DRY BAY
WET BAY
(FUEL)
DRY BAY
FUEL RIB
A
EB WELD
TIG WELG
A
TUBULAR TRUSS SPAR
SECTION A–A

Figure 16-7. Structural Details - Monocoque (Welded) Wing Design

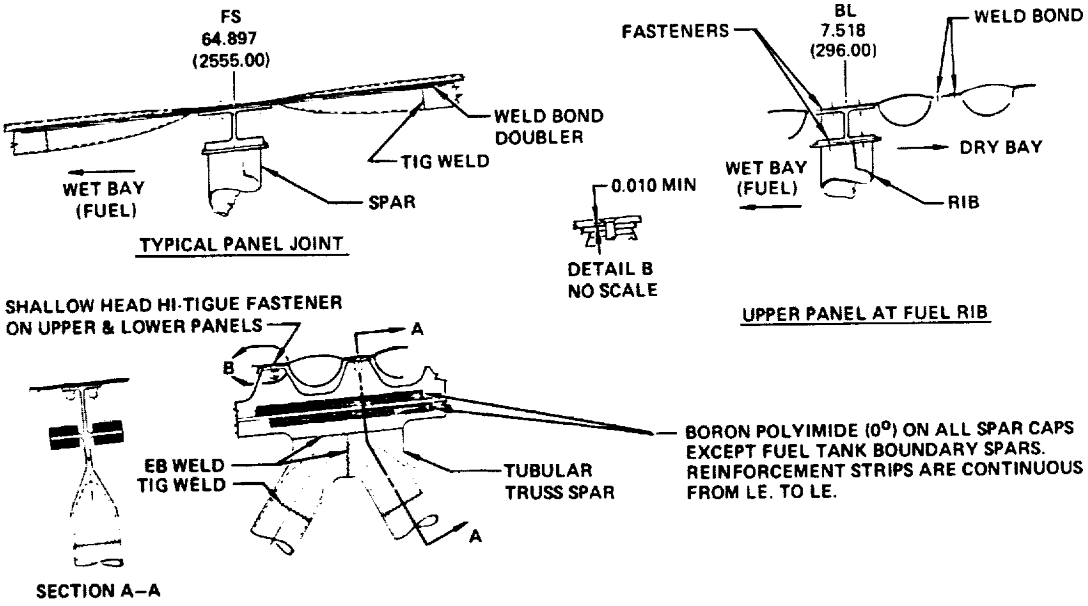

FS
64.897
(2555.00)
WELD BOND
DOUBLER
TIG WELD
SPAR
WET BAY
(FUEL)
TYPICAL PANEL JOINT
FASTENERS
BL
7.518
(296.00)
WELD BOND
DRY BAY
WET BAY
(FUEL)
RIB
0.010 MIN
DETAIL B
NO SCALE
UPPER PANEL AT FUEL RIB
SHALLOW HEAD HI-TIGUE FASTENER
ON UPPER & LOWER PANELS
B
A
EB WELD
TIG WELD
TUBULAR
TRUSS SPAR
A
BORON POLYIMIDE (0°) ON ALL SPAR CAPS
EXCEPT FUEL TANK BOUNDARY SPARS.
REINFORCEMENT STRIPS ARE CONTINUOUS
FROM LE. TO LE.
SECTION A–A

Figure 16-8. Structural Details - Composite Reinforced Wing Design

COST RESULTS

Table 16-1 summarizes the total manhours, material costs and tool-make manhours for the forward, aft and outer wing areas to manufacture the first production aircraft. The cost for the honeycomb core surface panels are included as Material Dollars since they are considered as a purchased item.

Tables 16-2 through 16-6 presents the production costs for each wing design. The details of the forward wing area are provided for each design. The methodology used to adapt the total production manhours, tool-make manhours and material costs to "value per pound" increments for input into the ASSET computer program is also presented The summary of production costs in terms of "value per pound" are tabulated in Table 16-7.

TABLE 16-1. TOTAL MANHOUR, MATERIAL AND TOOLING COSTS

ITEM	UNITS	DESIGN CONCEPTS				
		1	2	3	4	5
FABRICATION AND SUB ASSEMBLY MANHOURS	MANHOUR x 10^3	456	394	701	717	439
JOINT ASSEMBLY AND TANK SEAL MANHOURS	MANHOUR x 10^3	308	716	1623	90	354
TOTAL MANHOURS	MANHOUR x 10^3	764	1140	2324	807	793
MATERIAL COST	\$ x 10^3	1566	1881	3982	4033	2432
TOOL-MAKE MANHOURS	MANHOUR x 10^3	8024	5127	6249	6117	8058
WEIGHT	LB x 10^3	71	69	57	60	54

TABLE 16-2. PRODUCTION COSTS - CHORDWISE STIFFENED WING DESIGN

FORWARD WING AREA	OPERA. / ITEM	VALUE PER FOOT ÷	WEIGHT PER FOOT =	VALUE PER LB. ×	TOTAL WEIGHT =	TOTAL FAB. MANHRS. =	TOTAL MATERIAL DOLLARS =	TOTAL TOOLING MANHRS.
UPPER SKIN	LABOR	1.013	0.825	1.228	4,726	5,804	–	–
	MATERIAL	–	0.825	25.000	4,726	–	118,150	–
	TOOLING	142.871	0.825	173.177	4,726	–	–	818,435
LOWER SKIN	LABOR	1.242	0.942	1.319	4,726	6,234	–	–
	MATERIAL	–	0.942	25.000	4,726	–	118,150	–
	TOOLING	142.871	0.942	151.668	4,726	–	–	716,783
TRUSS SPARS	LABOR	45.349	2.597	17.462	7,954	138,893	–	–
	MATERIAL	69.936	2.597	26.930	7,954	–	214,201	–
	TOOLING	273.801	2.597	105.430	7,954	–	–	838,590
CIRCULAR - ARC SPARS	LABOR	40.760	3.886	10.489	2,114	22,174	–	–
	MATERIAL	98.281	3.886	25.291	2,114	–	53,465	–
	TOOLING	400.000	3.886	102.934	2,114	–	–	217,603
TRUSS RIBS	LABOR	53.220	1.368	38,904	2,570	99,983	–	–
	MATERIAL	43.840	1.368	32.047	2,570	–	82,361	–
	TOOLING	182.615	1.368	133.490	2,570	–	–	343,069
Σ	FORWARD WING AREA					273,088	586,327	2,934,480
Σ	AFT AREA WING					136,768	651,933	1,494,318
Σ	OUTER WING AREA	CALCULATED AS ABOVE				45,825	247,470	968,517
Σ	ADDITIVES							
Σ	FABRICATION AND SUBASSEMBLY					455,681	1,485,730	5,397,315
Σ	ASSEMBLY AND TANK SEAL	CALCULATED AS ABOVE				308,403	80,230	2,627,000
Σ	TOTAL PER AIRCRAFT				70,589	764,084	1,565,960	8,024,315
Σ	VALUE PER POUND					10.824 M/H	$ 22.184	113.677 M/H

TABLE 16-3. PRODUCTION COSTS - SPANWISE STIFFENED WING DESIGN

FORWARD WING AREA	OPERA. / ITEM	VALUE PER FOOT	÷ WEIGHT PER FOOT	= VALUE PER LB.	× TOTAL WEIGHT	= TOTAL FAB. MANHRS	= TOTAL MATERIAL DOLLARS	= TOTAL TOOLING MANHRS.
UPPER SKIN	LABOR	2.822	1.514	1.864	7,921	14,765		
	MATERIAL	37.850	1.514	25.000	7,921		198,025	
	TOOLING	44.480	1.514	29.379	7,921			232,711
LOWER SKIN	LABOR	3.412	1.181	2.890	7,921	22,884		
	MATERIAL	29.525	1.181	25.000	7,921		198,025	
	TOOLING	43.722	1.181	37.021	7,921			293,243
TRUSS SPARS	LABOR	31.885	1.238	25.755	2,387	61,477		
	MATERIAL	36.200	1.238	29.240	2,387		69,796	
	TOOLING	222.321	1.238	179.581	2,387			428.660
CIRCULAR ARC SPARS	LABOR	30.569	4.974	6.146	1,526	9,379		
	MATERIAL	114.870	4.974	23.090	1,526		35,235	
	TOOLING	369.643	4.974	74.315	1,526			113,405
TRUSS RIBS	LABOR	34.328	1.250	27.462	4,880	134,015		
	MATERIAL	45.520	1.250	36.420	4,880		177,681	
	TOOLING	89.399	1.250	71.519	4,880			349,013
CIRCULAR - ARC RIBS	LABOR	7.665	2.783	2.754	729	2,008		
	MATERIAL	86.800	2.783	31,190	729		22,738	
	TOOLING	56.820	2.783	20.417	729			14,884
Σ	FORWARD WING AREA					244,528	701,500	1,431,916
Σ	AFT WING AREA					73,971	631,398	443,033
Σ	OUTER WING AREA	CALCULATED AS ABOVE				27,845	315,018	335,402
Σ	ADDITIVES					47,977	154,119	361,102
Σ	FABRICATION AND SUBASSEMBLY					394,321	1,802,035	2,571,453
Σ	ASSEMBLY AND TANK SEAL	CALCULATED AS ABOVE				716,000	78,866	2,556,000
Σ	TOTAL PER AIRCRAFT				69,412	1,110,321	1,880,901	5,127,453
Σ	VALUE PER POUND					15.996 M/H	$27.097	73.869 M/H

TABLE 16-4. PRODUCTION COSTS - MONOCOQUE WING DESIGN

FORWARD WING AREA	OPERA. / ITEM	VALUE PER FOOT	÷ WEIGHT PER FOOT	= VALUE PER LB.	× TOTAL WEIGHT	= TOTAL FAB. MANHRS.	= TOTAL MATERIAL DOLLARS	= TOTAL TOOLING MANHRS.
UPPER SKIN	LABOR	24.248	1.764	13.746	7,255	99,735		
	MATERIAL	124.803	1.764	70.750	7,255		513,291	
	TOOLING	9.414	1.764	5.337	7,255			38,720
LOWER SKIN	LABOR	24.248	1.799	13.479	7,401	99,758		
	MATERIAL	113.895	1.799	63.310	7,401		468,557	
	TOOLING	9.414	1.799	5.233	7,401			38,729
TRUSS SPARS	LABOR	31.268	0.876	35.694	4,616	164.764		
	MATERIAL	28.508	0.876	32.543	4,616		150,219	
	TOOLING	279.545	0.876	319.115	4,616			1,473,035
CIRCULAR - ARC RIBS	LABOR	9.176	2.163	4.242	2,710	11,496		
	MATERIAL	47.744	2.163	22.073	2,710		61,444	
	TOOLING	75.891	2.163	35.085	2,710			95,080
Σ	FORWARD WING AREA					375,753	1,193,511	1,645,564
Σ	AFT WING AREA					222,845	1,544,077	1,394,018
Σ	OUTER WING AREA	CALCULATED AS ABOVE				51,610	1,042,782	123,880
Σ	ADDITIVES					50,412	137,170	530,048
Σ	FABRICATION AND SUBASSEMBLY					700,620	3,917,540	3,693,510
Σ	ASSEMBLY AND TANK SEAL	CALCULATED AS ABOVE				1,622,959	64,661	2,600,000
Σ	TOTAL PER AIRCRAFT				56,908	2,323,579	3,982,201	6,293,510
Σ	VALUE PER POUND					40.830	69.976	110.591

TABLE 16-5. PRODUCTION COSTS - MONOCOQUE (WELDED) WING DESIGN

FORWARD WING AREA	OPERA. / ITEM	VALUE PER FOOT ÷	WEIGHT PER FOOT =	VALUE PER LB ×	TOTAL WEIGHT =	TOTAL FAB. MANHRS. =	TOTAL MATERIAL DOLLARS =	TOTAL TOOLING MANHRS.
UPPER SKIN	LABOR	32.330	1.764	18.327	7,255	132,962		
	MATERIAL	124.803	1.764	70.750	7,255		513,291	
	TOOLING	27.426	1.764	15.548	7,255			112,801
LOWER SKIN	LABOR	34.126	1.799	18.969	7,401	140,390		
	MATERIAL	113.895	1.799	63.310	7,401		468,557	
	TOOLING	27.426	1.799	15.245	7,401			112,828
TRUSS SPARS	LABOR	17.773	0.876	20.288	5,965	121,018		
	MATERIAL	28.508	0.876	32.543	5,965		194,119	
	TOOLING	52.272	0.876	59.671	5,965			355,938
CIRCULAR - ARC RIBS	LABOR	9.176	2.163	4.242	3,706	15,721		
	MATERIAL	49.042	2.163	22.673	3,706		84,026	
	TOOLING	75.891	2.163	35.085	3,706			130,025
Σ	FORWARD WING AREA					410,091	1,259,993	711,592
Σ	AFT WING AREA					171,652	1,586,825	391,810
Σ	OUTER WING AREA		CALCULATED AS ABOVE			84,715	1,048,537	583,724
Σ	ADDITIVES					50,412	137,170	530,048
Σ	FABRICATION AND SUBASSEMBLY					716,870	4,032,525	2,217,174
Σ	ASSEMBLY AND TANK SEAL		CALCULATED AS ABOVE			90,336	–	3,900,000
Σ	TOTAL PER AIRCRAFT				60,104	807,206	4,032,525	6,117,174
Σ	VALUE PER POUND					13.430 M/H	67.092 $	101.776 M/H

TABLE 16-6. PRODUCTION COSTS - COMPOSITE REINFORCED WING DESIGN

FORWARD WING AREA	OPERA. ITEM	VALUE PER FOOT +	WEIGHT PER FOOT =	VALUE PER LB ×	TOTAL WEIGHT =	TOTAL FAB. MANHRS. =	TOTAL MATERIAL DOLLARS =	TOTAL TOOLING MANHRS.
UPPER SKIN	LABOR MATERIAL TOOLING	1.013 40.225 142.871	1.609 1.609 1.609	0.629 25.000 88.795	3,651 3,651 3,651	2,297	91,275	324,191
LOWER SKIN	LABOR MATERIAL TOOLING	1.242 33.375 142.871	1.335 1.335 1.335	0.930 25.000 107.020	3,651 3,651 3,651	3,395	91,275	390,730
TRUSS SPARS	LABOR MATERIAL TOOLING	48.967 591.900 342.251	6.380 6.380 6.380	7.675 92.770 53,644	7,883 7,883 7,883	60,502	731,306	422,876
CIRCULAR-ARC SPARS	LABOR MATERIAL TOOLING	44.378 578.290 500.000	7.574 7.574 7.574	5.859 76.350 66.015	4,485 4,485 4,485	26,278	342,430	296,077
TRUSS RIBS	LABOR MATERIAL TOOLING	53,220 163.000 182.615	3.597 3.597 3.597	14.796 45.316 50.769	2,214 2,214 2,214	32,758	100,330	112,403
CIRCULAR-ARC RIBS	LABOR MATERIAL TOOLING	36.863 74.485 249.293	2.865 2.865 2.865	12.867 25.999 87.013	1,430 1,430 1,430	18,400	37,179	124,429
Σ	FORWARD WING AREA					143,630	1,393,795	1,670,706
Σ	AFT WING AREA					248,931	546,183	2,776,071
Σ	OUTER WING AREA	CALCULATED AS ABOVE				46,454	430,149	1,011,601
Σ	ADDITIVES							
Σ	FABRICATION AND SUBASSEMBLY					439,015	2,370,127	5,458,378
Σ	ASSEMBLY AND TANK SEAL	CALCULATED AS ABOVE				354,001	61,366	2,627,000
Σ	TOTAL PER AIRCRAFT				54,013	793,016	2,431,493	8,085,378
Σ	VALUE PER POUND					14.682 M/H	45.017 $	149.693 M/H

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TABLE 16-7. SUMMARY OF PRODUCTION COSTS

VALUE PER POUND	DESIGN CONCEPTS				
	1	2	3	4	5
PRODUCTION MANHOURS/LB	10.8	16.0	40.8	13.4	14.7
MATERIAL $/LB	22.2	27.1	70.0	67.1	45.0
TOOL MAKE MANHOURS/LB	114.0	73.9	109.6	101.8	149.7

SECTION 17

CONCEPT EVALUATION AND SELECTION

BY

I. F. SAKATA

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SECTION 17

CONCEPTS EVALUATION AND SELECTION

INTRODUCTION

Analytical Design Studies (Task I) were performed to assess the relative merits of the various structural arrangements, concepts and materials applicable to the arrow-wing supersonic transport configuration defined in Section 2, Baseline Configuration Concept.

A spectrum of structural approaches for wing and fuselage applications that fully exploit the practically attainable advantages of near-term structures and materials technology were evaluated. Both smooth-skin and beaded-skin designs were explored considering advanced producibility techniques available for design of a near-term supersonic cruise aircraft.

The results of the detailed structural analysis were used to develop preliminary design drawings for mass and cost estimation. These data were then used for simplified cost benefit studies to evaluate the relative merits of each structural approach and identify those concepts which would merit further detailed engineering design and analysis (Task II).

EVALUATION CRITERIA

The criteria for evaluation of the structural design concepts include measures of: structural mass; mission capability; airplane size; development, production (material and fabrication), operational costs; and technology risk.

EVALUATION PROCEDURE

The evaluation process is performed in two steps:

1. The structural mass of the aircraft is estimated for each of the candidate structural approaches based on the premise of a fixed vehicle size and taxi mass (340,000 kilograms). This permits the determination of the allowable fuel for the aircraft and hence its range capability. RDT&E, production and maintenance costs. for each of the candidate structural approaches are then determined. A direct comparison of the structural mass, range and cost is made on the basis of constant airplane configuration and gross mass. The main shortcoming of this approach is that, since all three factors of structural mass, range performance and cost are variables, one cannot establish a single criterion for the selection of the optimum concept. Thus, the purpose of this step is to establish the basis for the evaluation methodology that follows.

2. The airplane configuration and gross mass are resized to meet the payload (22,000 kg) and range (7,800 km) requirements. The purpose of the resizing is not to suggest that the airplane configuration (size) be changed, but rather to provide a tool for assessing the impact of the candidate structural concepts and materials evaluated on a common basis, i.e., constant payload/range performance. Once this is done, the concept evaluation and selection may be performed on the basis of minimum-total-system-cost (as tempered by technology availability and risk considerations).

   In this context, total-system-cost includes the effects of:

   - Structural efficiency/material properties
   - Design considerations such as fuel tank sealing and ease of fabrication and assembly
   - Varying thrust, engine size, and engine mass requirements as airplane size changes
   - Impact of the structural/material concept, airplane size, and mass on fuel consumption; development, manufacturing and operating (maintenance) costs.

DOC (Direct Operating Cost) is a measure of total-system-cost that includes all the parameters noted and is chosen as the evaluation criterion in this study.

In order to assess the relative importance of the characteristics of each structural approach on the major component of DOC and provide a comparison between the cost and risk factors, the results of the aforementioned evaluation procedure are presented in the form of cost-benefit tradeoffs. Typical tradeoffs shown on Figure 17-1 are:

- Structural mass versus cost
- Technology improvement versus cost
- Technical risk versus cost
- Investment versus cost

As noted above, the selection of the best concept is performed on the basis of minimum-total-system-cost to satisfy the given payload/range requirement at acceptable program technology risk levels.

The ASSET Vehicle Synthesis Model

The parametric sizing, and performance evaluation of the structural design concepts are performed through the use of the Lockheed developed ASSET (Advanced System Synthesis and Evaluation Technique) vehicle synthesis model. A schematic presentation of the primary input and output data involved in the ASSET Synthesis Cycle, is shown on Figure 17-2. The ASSET Program integrates input data describing vehicle geometry, aerodynamics, propulsion, structures/materials, weights, and subsystems, and determines candidate vehicles which satisfy given mission and payload requirements. It provides the means to assess the effects of design options (thrust/weight, wing loading, engine cycle, advanced materials usage, etc.) on the vehicle weight, size, and performance. The key elements and the flow of information through ASSET are depicted in Figure 17-3.

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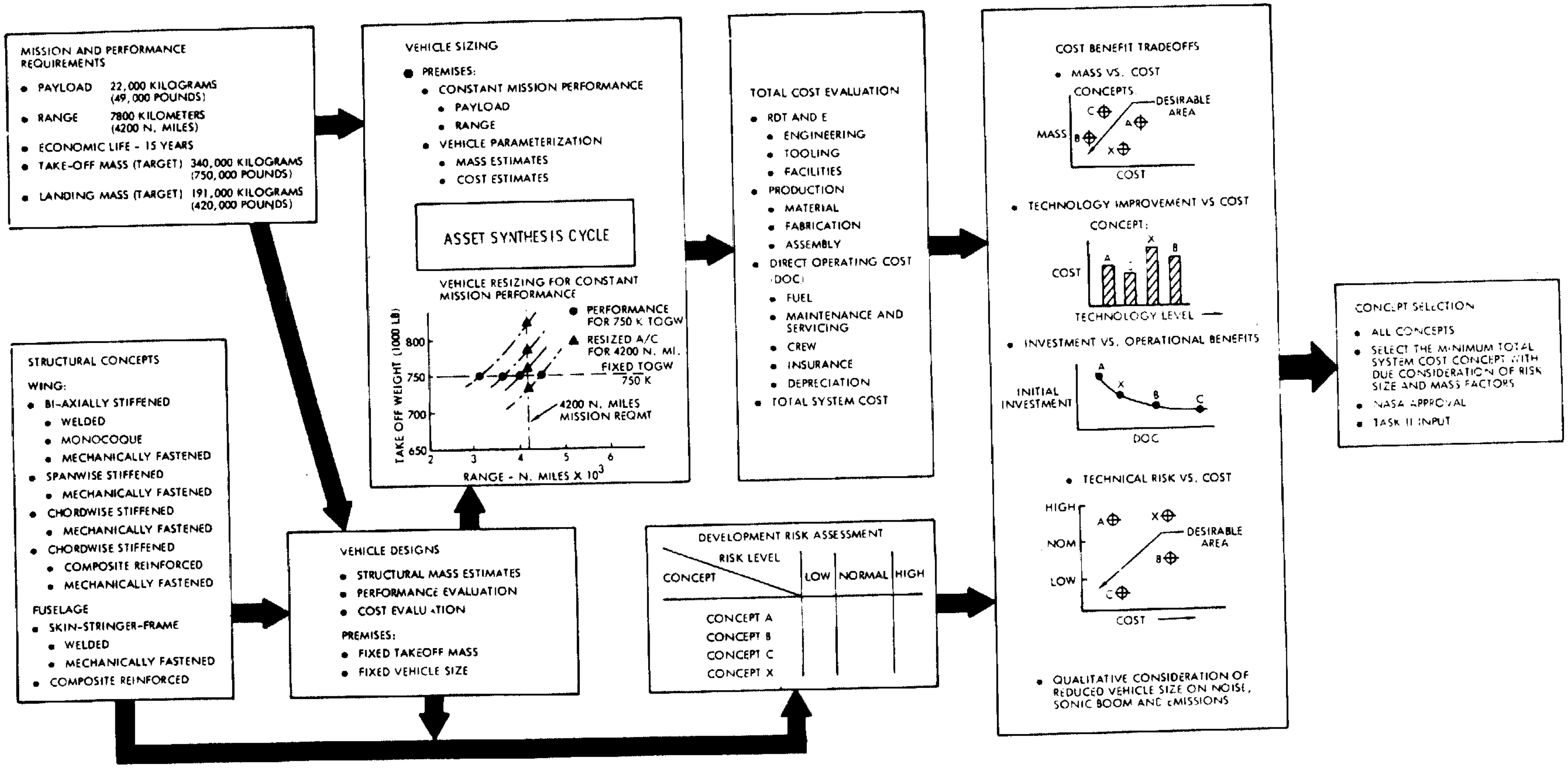
MISSION AND PERFORMANCE REQUIREMENTS
• PAYLOAD 22,000 KILOGRAMS (49,000 POUNDS)
• RANGE 7800 KILOMETERS (4200 N. MILES)
• ECONOMIC LIFE - 15 YEARS
• TAKE-OFF MASS (TARGET) 340,000 KILOGRAMS (750,000 POUNDS)
• LANDING MASS (TARGET) 191,000 KILOGRAMS (420,000 POUNDS)
STRUCTURAL CONCEPTS
WING:
• BI-AXIALLY STIFFENED
• WELDED
• MONOCOQUE
• MECHANICALLY FASTENED
• SPANWISE STIFFENED
• MECHANICALLY FASTENED
• CHORDWISE STIFFENED
• MECHANICALLY FASTENED
• CHORDWISE STIFFENED
• COMPOSITE REINFORCED
• MECHANICALLY FASTENED
FUSELAGE
• SKIN-STRINGER-FRAME
• WELDED
• MECHANICALLY FASTENED
• COMPOSITE REINFORCED
VEHICLE SIZING
• PREMISES:
• CONSTANT MISSION PERFORMANCE
• PAYLOAD
• RANGE
• VEHICLE PARAMETERIZATION
• MASS ESTIMATES
• COST ESTIMATES
ASSET SYNTHESIS CYCLE
VEHICLE RESIZING FOR CONSTANT MISSION PERFORMANCE
● PERFORMANCE FOR 750 K TOGW
▲ RESIZED A/C FOR 4200 N. MI.
FIXED TOGW 750 K
4200 N. MILES MISSION REQMT
TAKE OFF WEIGHT (1000 LB)
800
750
700
650
2
3
4
5
6
RANGE - N. MILES X 10³
VEHICLE DESIGNS
• STRUCTURAL MASS ESTIMATES
• PERFORMANCE EVALUATION
• COST EVALUATION
PREMISES:
• FIXED TAKEOFF MASS
• FIXED VEHICLE SIZE
TOTAL COST EVALUATION
• RDT AND E
• ENGINEERING
• TOOLING
• FACILITIES
• PRODUCTION
• MATERIAL
• FABRICATION
• ASSEMBLY
• DIRECT OPERATING COST (DOC)
• FUEL
• MAINTENANCE AND SERVICING
• CREW
• INSURANCE
• DEPRECIATION
• TOTAL SYSTEM COST
DEVELOPMENT RISK ASSESSMENT
RISK LEVEL
CONCEPT
LOW
NORMAL
HIGH
CONCEPT A
CONCEPT B
CONCEPT C
CONCEPT X
COST BENEFIT TRADEOFFS
• MASS VS. COST
CONCEPTS
DESIRABLE AREA
MASS
C
A
B
X
COST
• TECHNOLOGY IMPROVEMENT VS COST
CONCEPT:
A
X
B
COST
TECHNOLOGY LEVEL
• INVESTMENT VS. OPERATIONAL BENEFITS
INITIAL INVESTMENT
A
X
B
C
DOC
• TECHNICAL RISK VS. COST
HIGH
NOM
LOW
A
X
DESIRABLE AREA
B
C
COST
• QUALITATIVE CONSIDERATION OF REDUCED VEHICLE SIZE ON NOISE, SONIC BOOM AND EMISSIONS
CONCEPT SELECTION
• ALL CONCEPTS
• SELECT THE MINIMUM TOTAL SYSTEM COST CONCEPT WITH DUE CONSIDERATION OF RISK SIZE AND MASS FACTORS
• NASA APPROVAL
• TASK II INPUT

FIGURE 17-1. CONCEPT EVALUATION

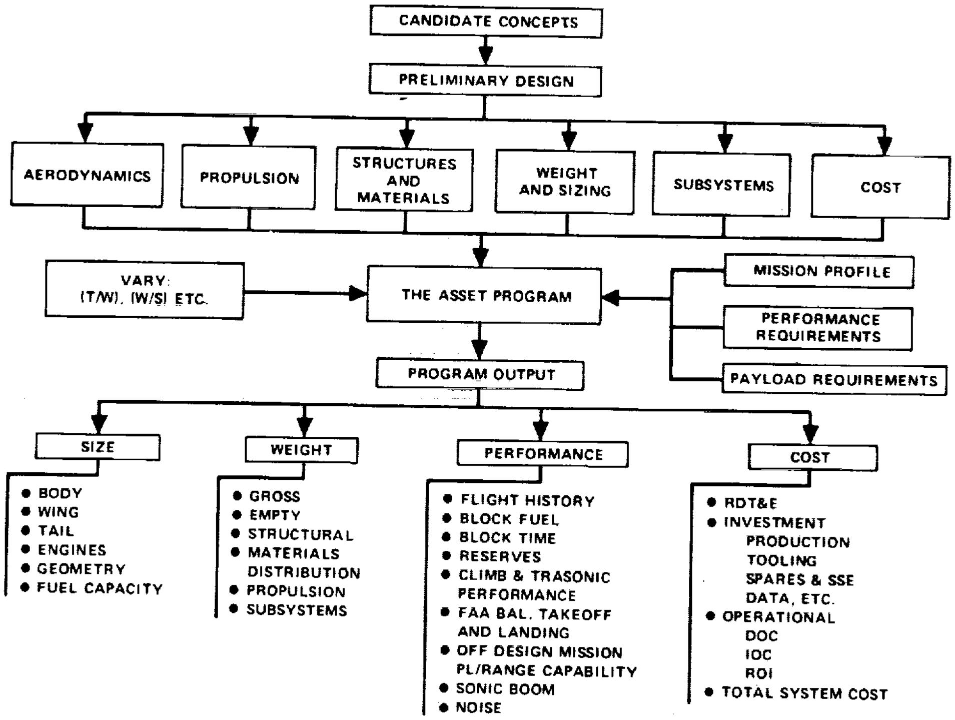
CANDIDATE CONCEPTS
PRELIMINARY DESIGN
AERODYNAMICS
PROPULSION
STRUCTURES AND MATERIALS
WEIGHT AND SIZING
SUBSYSTEMS
COST
VARY: (T/W), (W/S) ETC.
THE ASSET PROGRAM
MISSION PROFILE
PERFORMANCE REQUIREMENTS
PAYLOAD REQUIREMENTS
PROGRAM OUTPUT
SIZE
● BODY
● WING
● TAIL
● ENGINES
● GEOMETRY
● FUEL CAPACITY
WEIGHT
● GROSS
● EMPTY
● STRUCTURAL
● MATERIALS DISTRIBUTION
● PROPULSION
● SUBSYSTEMS
PERFORMANCE
● FLIGHT HISTORY
● BLOCK FUEL
● BLOCK TIME
● RESERVES
● CLIMB & TRASONIC PERFORMANCE
● FAA BAL. TAKEOFF AND LANDING
● OFF DESIGN MISSION PL/RANGE CAPABILITY
● SONIC BOOM
● NOISE
COST
● RDT&E
● INVESTMENT
PRODUCTION
TOOLING
SPARES & SSE
DATA, ETC.
● OPERATIONAL
DOC
IOC
ROI
● TOTAL SYSTEM COST

FIGURE 17-2. THE ASSET SYNTHESIS CYCLE

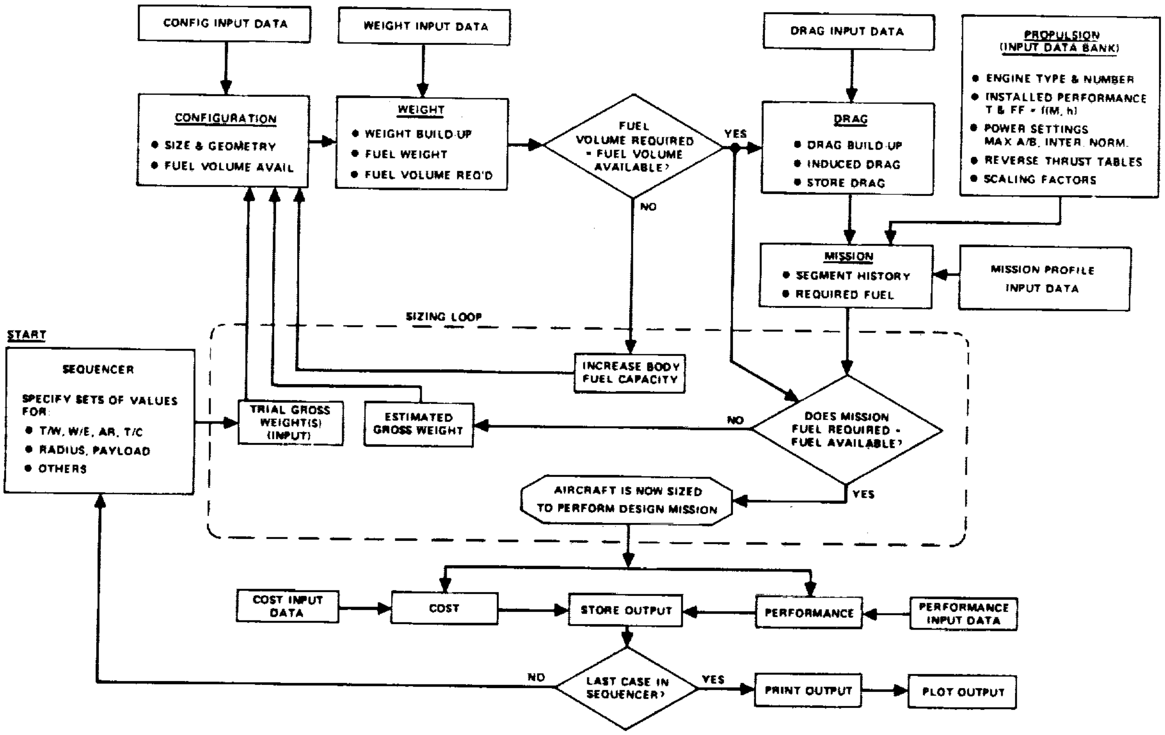
CONFIG INPUT DATA
WEIGHT INPUT DATA
DRAG INPUT DATA
PROPULSION (INPUT DATA BANK)
● ENGINE TYPE & NUMBER
● INSTALLED PERFORMANCE T & FF = f(M, h)
● POWER SETTINGS MAX A/B, INTER, NORM.
● REVERSE THRUST TABLES
● SCALING FACTORS
CONFIGURATION
● SIZE & GEOMETRY
● FUEL VOLUME AVAIL
WEIGHT
● WEIGHT BUILD-UP
● FUEL WEIGHT
● FUEL VOLUME REQ'D
FUEL VOLUME REQUIRED = FUEL VOLUME AVAILABLE?
YES
NO
DRAG
● DRAG BUILD-UP
● INDUCED DRAG
● STORE DRAG
MISSION
● SEGMENT HISTORY
● REQUIRED FUEL
MISSION PROFILE INPUT DATA
SIZING LOOP
START
SEQUENCER
SPECIFY SETS OF VALUES FOR:
● T/W, W/E, AR, T/C
● RADIUS, PAYLOAD
● OTHERS
TRIAL GROSS WEIGHT(S) (INPUT)
ESTIMATED GROSS WEIGHT
INCREASE BODY FUEL CAPACITY
NO
DOES MISSION FUEL REQUIRED = FUEL AVAILABLE?
YES
AIRCRAFT IS NOW SIZED TO PERFORM DESIGN MISSION
COST INPUT DATA
COST
STORE OUTPUT
PERFORMANCE
PERFORMANCE INPUT DATA
NO
LAST CASE IN SEQUENCER?
YES
PRINT OUTPUT
PLOT OUTPUT

FIGURE 17-3. ASSET PROGRAM SCHEMATIC

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The three major subprograms of ASSET are sizing, performance, and costing. The sizing subprogram sizes each parametric aircraft to a design mission. The design characteristics and component weights of the sized aircraft are then transferred to (1) the costing subprogram, which computes aircraft cost on the basis of component weights and materials, engine cycle and size, avionics packages, payload, production and operational schedules, and input cost factors, and (2) the performance subprogram which computes maximum speed, ceiling, landing and take-off distances and other performance parameters.

ASSET program output consists of a group weight statement, vehicle geometry description, mission profile summary, a summary of the vehicle's performance evaluation, and RDT&E, production and operational cost breakdowns (reference Appendix A).

Vehicle Sizing. - The sizing subprogram is composed of five routines: sequence, configuration, weight, drag, and mission. In addition, the sizing subprogram uses propulsion data input in the form of thrust and fuel flow tables and an independent atmosphere subroutine.

The sequence routine groups the sets of independent variables (design options and mission requirements) that are to be varied parametrically. Examples of these variables include (but are not limited to) thrust/weight, wing loading, aspect ratio, wing thickness ratio, wing sweep angle, design load factor, payload, equipment, avionics weights and volumes, materials usage factors, and design mission requirements (range, radius, endurance, speed, etc.).

The configuration routine computes the geometric data for the vehicle components (planform areas, wetted areas, frontal areas, lengths, diameters, chords, reference lengths, volumes, shapes, etc.) required by the weight and drag routines. The weight routine determines the component weight build-up, materials usage for the major airframe elements based on an input percentage distribution matrix and the fuel available. These data are utilized in the configuration routine. The configuration and weight routines, operating together, determine the geometric and weight characteristics for an airplane having an assumed trial takeoff gross weight. The trial vehicle is geometrically sized to contain the crew, equipment, payload, propulsion system and fuel. The tails are sized to provide specified (input) tail volume coefficients.

The drag routine constructs a drag build-up composed of friction drag, zero-lift pressure drag, and induced drag. Friction drag is determined for each vehicle component over a range of Mach numbers and altitudes using the component wetted areas, reference lengths, and roughness and from drag factors. The zero-lift pressure drag for the wing is computed over the Mach number range by the drag subroutine using the wing geometry characteristics (sweep, thickness, aspect ratio, leading edge radius). The zero-lift pressure drags for other vehicle components are based on the component areas, determined by the configuration routine, and the input values of the component zero-lift pressure drag coefficients. Induced drag (including the effects of compressibility) of the trial aircraft, based on the wing geometry, is determined as a function of Mach number and lift coefficient by the drag routine. The computed drag build-up for the aircraft and propulsion data for the engine under study are input to the program. Applicable power setting (takeoff, maximum, intermediate, maximum continuous, etc.) thrust and fuel flow data are provided as functions of Mach number and altitude. Partial power tables are used to simulate operation at thrust levels required during cruise or loiter.

The mission subroutine determines the fuel required to perform the design mission profile. The mission profile is assembled from specified flight segments, such as takeoff, climb, acceleration, cruise, loiter, etc. Simplified two dimensional point mass flight equations are used in determining the time history of the mission. Climbs follow predetermined speed-altitude schedules. Cruise and loiter segments may be performed at specified altitude and speed flight conditions, or the speed and/or the altitude can be optimized for maximum cruise range of maximum loiter endurance. Allowances are made for taxi, warmup and takeoff, landing, and fuel reserves.

An iterative convergence technique completes the sizing subprograms. The fuel available from the weight routine and the fuel required determined by the mission routine are compared. The iteration computes and passes new trial aircraft through the sizing cycle until acceptable agreement is reached between the

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available and required fuel. At this point the vehicle is properly sized to perform the specified design mission.

Performance Evaluation. - Any or all of the following vehicle performance capabilities can be evaluated: Climb, maximum speed, maneuverability, airport performance, and alternate mission capability. The climb characteristics are assessed at specified vehicle weights for given thrust settings. The maximum rate of climb at sea level is determined at the takeoff weight for a zero-acceleration climb schedule. Ceiling altitudes are determined for specified rate of climb requirements for a series of aircraft weights ranging from the takeoff weight to the zero fuel weight. Service and cruise ceilings may be determined by specification of the appropriate thrust settings, and rate of climb requirements.

Speed characteristics are assessed for specified aircraft weights and thrust settings. The maximum speed at sea level, the maximum speed at the optimum altitude, and the corresponding optimum altitude are determined.

Airport performance is evaluated for standard or non-standard days. Aerodynamic data representing the maximum lift coefficient and drag polars for the aircraft in the takeoff and landing configurations are provided by input. The distance required to take off over a 50 foot obstacle is determined for defined thrust settings. Takeoff and transition speeds are specified as percentages of the stall speed. Landing distances over a 50-foot obstacle may be determined for both flared and unflared approaches. Approach and touchdown speed are specified as percentages of the stall speed. Sinking speeds at the 50-foot height and at touchdown are constrained below defined limits. Thrust reversal may be employed during the braking phase. Go-around rate of climb during the landing approach is computed for specified thrust settings. Any number of engines may be inoperative.

Alternate mission off-design performance for the synthesized aircraft is determined by the mission routine. The basic difference between the mission routine of sizing subprograms and the alternate mission performance evaluation is that the former determines the fuel required to perform the design mission, whereas the latter determines the mission capabilities (range, radius, endurance, etc.) with the fuel available in the sized aircraft. The alternate mission profiles are assembled from specified flight segments (takeoff, climb, acceleration, cruise, loiter, etc.) in the same manner that the design (sizing) mission is constructed. However, one of the mission segments of the profile is a variable segment. Through an iterative convergence technique, the duration of the variable segment (distance, time, etc.) is adjusted so that the fuel required to perform the mission is equal to the onboard fuel. Payload and equipment weights, and fuel loads for the alternate missions are specified and may be different from those corresponding to the design mission.

Costing. - The costing program computes RDT&E, investment, and operational costs. Both the RDT&E and production (flyaway) aircraft costs are broken down by airframe, engines, avionics, and armament. Airframe costs are further broken down into engineering, tooling, manufacturing, quality control, and material costs. The various cost elements are computed on the basis of cost estimating relationships (CER) which are established by analysis of historical data of applicable aircraft programs, Lockheed's R&D and production experience, and subcontractor/supplier quotations. Cost input consists of dollars-per-hour (labor cost) and dollars-per-pound (material cost) factors by aircraft structural element and material, labor rates, production rates and schedule, learning curves, subsystem, engine and avionics cost factors, and operational (fuel, attrition, etc.) considerations. The model permits parametric costing as function of thrust, inert weight element and advanced material usage.

Cost Model Description

The cost models used in the evaluation of the arrow-wing configuration supersonic transport consists of subroutines to the ASSET program.

Development Cost Model. - The cost estimates for the primary elements of development cost are determined by cost estimating relationships (CER's) which are determined by statistical analysis of historical data from military programs. The basic equations used to estimate the development cost for the airframe and engine are modified versions of the CER's developed by the RAND Corporation (references 1

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and 2). The RAND equations are modified to reflect airframe and engine manufacturer's experience. The airframe engineering hour estimates by the RAND CER's are modified to reflect a Lockheed in-house estimate. The Lockheed estimate is provided by a methodology that has been developed through a detailed analysis of Lockheed programs. The modifications to the RAND equations are provided by the application of K factors to the basic equations.

The development cost model includes the following elements:

- Prototype Aircraft
- Design Engineering
- Development Test Articles
- Flight Test
- Engine Development
- Development Tooling
- Special Support Equipment
- Development Spares
- Technical Data
- Avionics Development

The equations for determining the cost for each of the above elements are shown in the Development Cost Model that follows.

The cost for the prototype aircraft is determined from the flyaway cost model and input to the development model. The prototype aircraft are costed on the basis of the first few vehicles produced.

Development Cost Model

Prototype Aircraft

TPROT = TFLCO * XNYO

Design Engineering

RFDE = 0.0396 * WAMPR ** 0.791 * SS ** 1.526 * CXNYO ** 0.183

DIH = RFDE * XKE - SELHO = (Design Engineering hours less sustaining)

DIEC = DIH (DER + OER) * (1 + APRFF) = (design engineering cost)

Tooling

DTHB = 4.0127 * WAMPR ** 0.764 * SS ** 0.899 * CXNYO ** 0.178 * DRT * 0.066

DHT = DTHB * XKT - PTLHO = (Design tooling hours less sustaining)

DTC = DTH (DTR + OTR) * (1 + APRFF) = (Design tooling cost)

Development Test Articles

DSTA = (TAFCO/CXNYO) * XNSTA = (Cost for static test article)

DFTA = (TAFCO/CXNYO) * XNFTA = (Cost for fatigue test article)

DMTS = (TAFCO/CXNYO) * XMTSF = (Cost for systems test articles)

DART = (DSTA + DFTA + DMTS) * (1 + APRFF)

Flight Test

RFFT = 0.001244 * WAMPR ** 1.16 * SS ** 1.371 * CXNYO ** 1.281

DFT = RFFT (1 + APRFF) * XKFT = (Flight test cost including profit)

Engine Development

CEDCM = XMMAX ** 0.62 [(CXNY10 + CXNYO) * XNENGC] ** 0.10

DCENG = CEDCF * [(TCE/1000)/XNENGC] ** CEDCE * CEDCM

Avionics

DAV = DPAVD * WAVUN + FAVDC

Spares

DSPAR = ADSF * TAFCO + EDSF * TENCO + AVDSF * TAVCO

Special Support Equipment

DSSE = DSSEF * TFLCO

Technical Data

$$DDATA = DTDF\ (TFLCO + DIEC + DTC + DART + DFT + DCENG + DLENG + DAV + DSPAR + DSSE + DOT + DMT)$$

The description of the inputs and the factors for the development model are included along with the description of the inputs for the production model that follows.

Investment Cost Model. - The Investment Cost Models includes subroutines to provide the cost for the aircraft, the aircraft spares, and the special support equipment. The primary element of investment is the aircraft and it is given the most attention in terms of detail and consideration of the labor and material cost factors. The spares and special support equipment cost are treated as percentages of the flyaway cost of the aircraft. The production cost estimate is made to the same general level of detail as the airplane group weight statement. The production cost input format includes the following elements:

- Material Cost Factors
- Labor Cost Factors
- Labor Rates
- Sizing and Learning Curve Factors
- Sustaining Engineering
- Sustaining Tooling
- Engineering Change Orders
- Quality Assurance.
- Miscellaneous Costs
- Warranty
- Insurance and Taxes
- Profit

An illustrative example of the elements of the airframe and their representative cost factors is shown in Table 17-1. How these factors are applied is illustrated in the schematic of the flyaway cost model shown in Figure 17-4.

Airframe Material Cost - As shown by Table 17-1, the material cost factors include representative cost factors for various types of material for the structural elements of the airframe. The airframe production cost model has space for material cost factor inputs for aluminum, titanium, steel, composites, and other. The various types of materials are listed across the top of the input sheet (Table 17-1). A material cost factor is assigned to each type

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TABLE 17-1
ASSET COST INPUT
COMPOSITE REINFORCED WING DESIGN

Flags and Misc Inputs	CSTPRT	0	FAST	1.0										1
	XMMAX	2.7	XMMIN	0.25										
	XNENGL	0	TMAXLE	0	XNAVS	1.0	TMR	1.73						3
Production Schedule (300 A/C)	XNY1	12	XNY2	18	XNY3	24	XNY4	30	XNY5	36	XNY6	36		
	XNY7	36	XNY8	36	XNY9	36	XNY10	36	XNY0	4				5
Material Type	AL		TI		Steel		Composite		Other					
Material Cost Factors ~ $/lb	DPWIAL	12.72	DPWITI	49.92	DPWIST	15.90	DPWICØ	114.48	DPWIØT	12.72			Wing	
	DPTAAL	12.72	DPTATI	38.90	DPTAST	15.90	DPTACØ	106.08	DPTAØT	12.72			Tail	
	DPFUAL	12.72	DPFUTI	25.55	DPFUST	15.90	DPFUCØ	114.48	DPFUØT	12.72			Fuselage	
	DPLGAL	37.10	DPLGTI	47.23	DPLGST	42.41	DPLGCØ	125.40	DPLGØT	37.10			Landing Gear	
	DPNAAL	15.90	DPNATI	21.35	DPNAST	19.88	DPNACØ	127.20	DPNAØT	15.90			Nacelle	
	DPAIAL	37.14	DPAITI	60.62	DPAIST	42.45	DPAICØ	109.56	DPAIØT	37.14			Air Induction	
	DPSCAL	127.20	DPSCTI	134.36	DPSCST	129.36	DPSCCØ	208.61	DPSCØT	127.20			Surface Controls	12
Labor Cost Factors ~ hr/lb	HPWIAL	4.80	HPWITI	7.581	HPWIST	7.20	HPWICØ	7.50	HPWIØT	4.80			Wing	
	HPTAAL	4.80	HPTATI	13.20	HPTAST	10.60	HPTACØ	7.50	HPTAØT	4.80			Tail	
	HPFUAL	6.00	HPFUTI	9.00	HPFUST	7.20	HPFUCØ	9.00	HPFUØT	6.00			Fuselage	
	HPLGAL	0.12	HPLGTI	0.12	HPLGST	0.12	HPLGCØ	0.12	HPLGØT	0.12			Landing Gear	
	HPNAAL	4.80	HPNATI	10.00	HPNAST	8.00	HPNACØ	6.00	HPNAØT	4.80			Nacelle	
	HPAIAL	5.40	HPAITI	11.20	HPAIST	9.00	HPAICØ	7.50	HPAIØT	5.40			Air Induction	
	HASCAL	4.80	HPSCTI	11.50	HPSCST	8.00	HPSCCØ	6.00	HPSCØT	4.80			Surface Controls	19
Non-Structural Element Cost Factors	DPEC	175.00	FXECMC		HPEC	2.50	FXECLH						Engine Controls	
	DPTR	0	FXTRMC		HPTR	0.14	FXTRLH						Thrust Reverser	
	DPES	51.60	FXESMC		HPES	2.58	FXESLH						ECS	
	DPXI	171.08	FXXIMC		HPXI	3.02	FXXILH						Instruments	
	DPHY	161.23	FXHYMC		HPHY	9.20	FXHYLH						Hydraulics	
	DPEL	95.43	FXELMC		HPEL	7.30	FXELLH						Electrical	
	DPESS	15.90	FXSSMC		HPESS	4.80	FXSSLH						Engine Section	
	DPFE	42.46	FXFEMC		HPFE	3.27	FXFELH						Furn. & Equip.	
	DPAM	0	FXAMMC		HPAM	0	FXAMLH						Armament	
	DPAG	44.54	FXAGMC		HPAG	0.12	FXAGLH						Auxiliary Gear	
	DPEI	0	FXEIMC		HPEI	0.14	FXEILH						Eng. Installation	
	DPAV	13.88	FXAVMC		HPAV	4.20	FXAVLH						Avionics Install.	
	DPSI	0.50	FXSIMC		HPSI	0.67	FXSILH						Systems Integr.	
	DPVC	0	FXVCMC		HPVC	0	FXVCLH						Vector Control	
	DPFS	126.61	FXFSMC		HPFS	4.35	FXFSLH						Fuel System	34
Manufacturing Support Factors	QAPO	0.20	ECPO	0	PTMPO	0.25	SEPO	0.15	RMRO	0.35	XMISCO	0.04		
	QAP	0.20	ECP	0	PTMP	0.12	SEP	0.10	RMR	0.35	XMISC	0.04		36
Eng & Avionics Prod Cost Factors	CEPCF	548,696	CEPCE	0.60	XLEPCF	0	XLEPCE	1.0	DPAVP	0	FAVPC	500,000		37
Tax Insurance, Warranty, and Profit Cost Factors	ATAIF	0.10	ETAIF	0	AVTAIF	0	AWAF	0.05	EWAF	0	AVWAF	0		
	APRFF	0.15	EPRFF	0	AVPRFF	0								39
Sizing and Learning Curve Factors	WEMTBM	227,064	XMCS	0.99										
	WEMTBL	227,064	XHCS	0.96										
	XNMB	100	XMLCS	0.95										
	XNHB	100	XHLCS	0.80										
	XNCEB	1.0	XCELCS	0.90										
	XNLEB	1.0	XLELCS	0.90										
	XNAB	1.0	XAVLCS	1.00										46

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TABLE 17-1

ASSET COST INPUT (Continued)

Material Type	AL		TI		Steel		Composite		Other				
Labor Rates	DER ØER DEGR ØEGR XMGR	8.17 9.20 0 0 0	DTR ØTR DTGR ØTGR XAGR	6.09 12.36 0 0 0	DMR ØMR DMGR ØMGR XEGR	5.12 10.72 0 0 0	DQAR ØQAR DQAGR ØQAGR	6.29 10.72 0 0	DMISC ØMISC	5.12 10.72			51
RDTE Cost Factors	XKE XNSTA CEDCF DRT DSSEF	0.72 1.0 21.3x10⁶ 1.0 0.02	XKT SNFTA CEDCE ADSF DTDF	0.86 1.0 0.55 0.15 0.005	XKFT XMTSF XLEDCE EDSF DØT	0.34 0.30 0 0.50 0	ETSMR XLEDCF AVDSF DMT	0 0 0.30 0	EFTMR DFAVD	0 0	FAVDC	0	56
Investment Cost Factors	PRT PSSEF	3.0 0.041	PECF PTDF	0 0.005	PECE PØT	1.0 0	APSF PMT	0.15 0	EPSF SCFM	0.30 1.05	AVPSF SCFL	0.30 1.05	58
Operational (Military) Cost Factors													65
Direct Operating Cost (DOC) Factors	XNRITE SSFB U	13.0 15.0 3600	YEAR IFB IRA	1973 20.0 0.0222	XNCREW FCSIR PERIOD	3.0 0.7 14.0	XNPASS MNTLIR CFT	234 0.05 0.02	XNATT TG COT	8.0 15.0 0.945	XLF MBF	0.55 1.90	68
Indirect Operating Cost (IOC) Factors	REVPAS XKSE XKPH	1.81x10⁶ .0206 12.0	AVCARG XKLØE XKCH	2000 2.87 96.0	XKCØ XKØP	58.0 .0061	XKAT XKØC	27.0 .0065	XKFB XKGA	0.58 0.064			71
Return on Investment (ROI) Factor	CFARE	.0496	XKFARE	9.0	XKFACC	0	TAXR	0.48	CARGF	0.30			72

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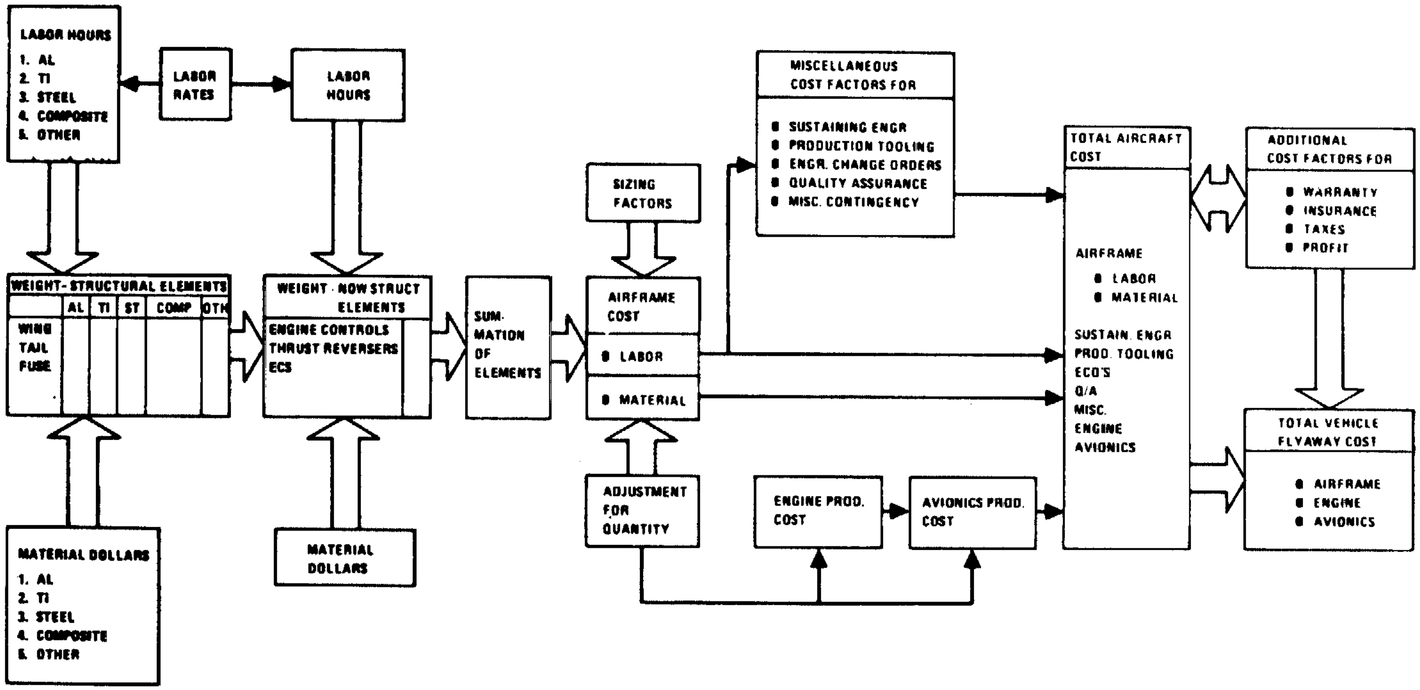
LABOR HOURS
1. AL
2. TI
3. STEEL
4. COMPOSITE
5. OTHER
LABOR RATES
LABOR HOURS
WEIGHT - STRUCTURAL ELEMENTS
AL
TI
ST
COMP
OTH
WING
TAIL
FUSE
WEIGHT - NON STRUCT ELEMENTS
ENGINE CONTROLS
THRUST REVERSERS
ECS
MATERIAL DOLLARS
1. AL
2. TI
3. STEEL
4. COMPOSITE
5. OTHER
MATERIAL DOLLARS
SUM-MATION OF ELEMENTS
SIZING FACTORS
AIRFRAME COST
• LABOR
• MATERIAL
ADJUSTMENT FOR QUANTITY
MISCELLANEOUS COST FACTORS FOR
• SUSTAINING ENGR
• PRODUCTION TOOLING
• ENGR. CHANGE ORDERS
• QUALITY ASSURANCE
• MISC. CONTINGENCY
ENGINE PROD. COST
AVIONICS PROD. COST
TOTAL AIRCRAFT COST
AIRFRAME
• LABOR
• MATERIAL
SUSTAIN. ENGR
PROD. TOOLING
ECO'S
Q/A
MISC.
ENGINE
AVIONICS
ADDITIONAL COST FACTORS FOR
• WARRANTY
• INSURANCE
• TAXES
• PROFIT
TOTAL VEHICLE FLYAWAY COST
• AIRFRAME
• ENGINE
• AVIONICS

FIGURE 17-4. AIRCRAFT FLYAWAY COST MODEL

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type of material as determined from the sizing program. The ASSET program determines the total weight of each element from the performance and configuration input data. After the total weight of the component is determined, the amount of each type of material is obtained by applying percentage factors to the total. The percentage factors for each type of material are established through previous analysis and input to the program.

Airframe Labor Cost - The same procedure as used in the materials is used in the labor subroutine, except that the labor is in hours. After the total number of hours are determined the labor rate is applied to arrive at the total labor cost.

The labor rates shown in Table 17-1 include the rates for design engineering, tooling, manufacturing, quality assurance, and miscellaneous. Only the labor rates for manufacturing and quality assurance are used for development engineering and tooling.

Non-Structural Elements Cost Factors - The cost factors for these elements includes both labor and material. This category includes the installation cost for the systems and equipment noted as well as their manufacturing cost with the exception of the engine and avionics. The installation costs for the engine and avionics are included here but the purchase costs for these items are shown separately.

After the labor hours, labor rates and material cost factors are applied to each material type, the elements are summed to arrive at a total airframe labor and material cost. These sums are then adjusted for quantity and size.

Sizing and Learning Curve Factors. - The sizing factors are included to account for scaling of the labor and material cost due to aircraft size. The learning curve factor accounts for cost change due to quantity produced. The labor and material cost factors shown in Table 17-1 are normalized to a particular vehicle weight and production quantity. The scaling factors modify the labor and material cost according to the size of the vehicle being analyzed and the number of aircraft in the production program. The sizing and learning curve factors include:

Material Sizing Factor
Labor Sizing Factor
Material Learning Curve
Labor Learning Curve
Engine Learning Curve
Avionics Learning Curve

As noted by Figure 17-4 the adjustment factors for quantity are applied to the engine and avionics as well as the labor and material.

Miscellaneous Factors - There are cost items which must be included in the production cost of the aircraft that are not part of the labor and material costs directly associated with the manufacturing of the vehicle. These are such items as quality assurance engineering changes, tool maintenance, sustaining engineering, warranty, taxes, insurance and miscellaneous costs. The costs for these items are added to the cost for the structural and non-structural elements to arrive at a total airframe cost. These factors are applied against the total airframe labor cost to arrive at the cost of each item. Costs are summed to obtain a total airframe cost.

Engine Cost - The engine cost estimate is provided by a production cost equation, or supplied by engine manufacturers, and input to the model. The equation is taken from the latest RAND revision (Reference 2) of their analysis of turbojet and turbofan production cost. The RAND equation has been modified by estimates provided by P&W and GE for the AST. The production cost equation for the duct burning turbofan is of the form:

$$\text{Engine Production Cost} = 631{,}000 \left(\frac{\text{TCE}}{1000}\right)^{0.6} \left(\text{XNENGC}\right)^{-0.152}$$

where

TCE = maximum sea level static thrust

XNENGC = number of engines in the production program.

The constant in the equation is changed to 546,000 for costing the turbojet engine.

Avionics - The avionics estimates are provided by vendors or in-house analysis and input to the model.

Additional Factors - The total summation of cost elements up to this point produces the flyaway cost of the aircraft without profit and costs for warranty, taxes, and insurance. The cost for these items is obtained by applying factors for each to the total aircraft cost. These costs are incorporated into the total aircraft production cost to arrive at the total vehicle flyaway cost except for the amortized R&D.

Development and Production Model Symbol Definitions

TFLCO = Cost of prototype

CSTRT = Print Indicator (1 = detail, 0 = summary)

FAST = Indicator if AST or other

XMMAX = Maximum Mach number

XMTN = Minimum Mach Number - Stall Speed

XNENGL = Number of lift engines

TMAXLE = Maximum thrust of lift engines

XNAVS = Number of avionics suites

TMR = Tooling material rate

XNY1-XNY = Number of aircraft delivered per year

XNYO = Number of aircraft in the development program

QAPO/QAP = Quality assurance factor for development/and production

ECPO/ECP = Engineering change order factor for development/and production

PTMPO/PTMP = Tool maintenance factor for development/and production

SEPO/SEP = Sustaining engineering factor for development/and production

RMRO/RMR = Raw material rate for development and production

XMISCO/XMISC = Miscellaneous cost factor for development/and production

CEPCF = Constant value for engine production cost formula - cruise engine

CEPCE = Value of coefficient in engine production cost formula - cruise engine

XLEPCF = Constant value for engine production cost formula - lift engines

XLEPCE = Value of coefficient in engine production cost formula - lift engine

DPAVP = Avionics production cost factor

FAVPC = Production cost for avionics

ATAIF = Airframe insurance factor

ESTAIF = Engine insurance factor

AVTAIF = Avionics insurance factor

AWAF = Airframe warranty factor

EWAF = Engine warranty factor

AVWAF = Avionics warranty factor

APRFF = Airframe profit factor

EPRFF = Engine profit factor

AVPRFF = Avionics profit factor

WEMTBM = Weight empty of aircraft being evaluated

WEMTBL = Weight empty of base line vehicle from which the cost factors were developed

XNMB = Quantity at which the material factors were developed

XNHB = Quantity at which the labor factors were developed

XNCEB = Base quantity for cruise engines

XNLEB = Base quantity for lift engines

XNAB = Base quantity for avionics

XMCS = Material cost sizing coefficient

XHCS = Labor cost sizing coefficient.

XMLCS = Material learning curve slope

XHLCS = Labor learning curve slope

XCELCS = Cruise engine learning curve slope

XLELCS = Lift engine learning curve slope

DER = Engineering labor rate (direct)

OER = Engineering overhead rate (indirect)

DEGR =
OEGR = Growth rates - not used
XMGR =

DTR = Tooling labor rate (direct)

OTR = Tooling overhead (indirect)

DTGR =
OTGR = Growth rates - not used
XAGR =

DMR = Manufacturing labor rate (direct)

OMR = Manufacturing overhead rate (indirect)

DMGR =
OMGR = Growth rates - not used
XEGR =

DQAR = Quality assurance labor rate (direct)

OQAR = Quality assurance overhead rate (indirect)

DQAGR = Growth rates - not used
OQAGR =

DMISC = Labor rate for miscellaneous items

OMISC = Overhead rate for miscellaneous items

XKE = Complexity factor for engineering

XKT = Complexity factor for tooling

XKFT = Complexity factor for flight test

XNSTA = Number of test articles for structural tests

XNFTA = Number of test articles for fatigue tests

XMTSF = Number of test articles for systems test

ETSMR = Engineering test material rate

EFTMR = Flight material rate

CEDCF = Constant value for cruise engine development cost equation

CEDCE = Value of coefficient for development cost formula for cruise engines

XLEDCF = Constant value for lift engine development equation

XLEDCE = Value of coefficient for development cost formula for lift engines

DPAVD = Development cost factor for avionics

FAVDC = Development cost for avionics

DRT = Production rate for development

ADSF = Airframe spares factor for development

EDSF = Engine spares factor for development

AVDSF = Avionics spares factor development

DSSEF = Special suppport cost factor for development

DTDF = Technical data cost factor for development

DOT = Operator trainer cost factor for development

DMT = Maintenance trainer cost factor for development

PRT = Maximum monthly production rate

PECF = Constant term for production engineering cost formula

PECE = Value of coefficient for production engineering cost formula

APSF = Spares factor for production airframes

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EPSF = Spares factor for production engines

AVPSF = Spares factor for production avionics

PSSEF = Special support equipment cost factor for production

PTDF = Technical data cost factor for production

POT = Cost for operator trainers

PMT = Cost for maintenance trainers

SCFM = Material cost factor for maintenance

SCFL = Labor cost factor for maintenance

Operating Cost Models. - The operating cost includes the standard elements normally found in the direct and indirect operating cost (DOC/IOC) as reported by the airlines. The DOC model is a modified version of the 1967 ATA method (Reference 3). The modifications to the DOC equations in the ATA method consists of: 1) combining the crew cost equations into a general expression for any number of crew members and 2) expanding the maintenance equations into greater detail. The more detailed maintenance equations are obtained from (Reference 4). The IOC model consists of set of expression derived through the combined efforts of Lockheed and Boeing (Reference 5). The indirect expense factors are those experienced by the international carriers (Reference 6).

DOC Model

Flight Crew $[3.0 * (45 + \mathrm{SSFB}) + 35\,(\mathrm{XNCREW} - 3) + \mathrm{IFB}] * (1.0 + \mathrm{FCSIR})^{\mathrm{XNYR}} * \mathrm{U}$

Fuel and Oil $1.02 * \mathrm{U} * (\mathrm{FB}/\mathrm{TB} * \mathrm{CFT} + \mathrm{XNENGC} * \mathrm{COT} * 0.135)$

Insurance $\mathrm{IRA} * \mathrm{TUACC}$

Depreciation $(\mathrm{TUACC} + \mathrm{SPARES})/\mathrm{PERIOD}$

Maintenance Equipment and Furnishings

Labor $= [0.5\mathrm{TF} + 1.0 + (4.5\mathrm{TF} + 18) * \mathrm{WAF}/10^{6}] * \mathrm{U}/\mathrm{TB} * \mathrm{MNTLR}$

Material $= [0.4\mathrm{TF} + 1.20 + (14\mathrm{TF} + 42) * \mathrm{WAF}/10^{6}] * \mathrm{U}/\mathrm{TB}$

Landing Gear

$$\text{Labor} = (1.0 + 10 * \text{WAF}/10^{6}) * \text{U/TB} * \text{MNTLR}$$

$$\text{Material} = (2.4 + 1.50 * \text{TUAFC}/10^{6}) * \text{U/TB}$$

Tires and Brakes

$$\text{Material} = (1.2 + 7.0 * \text{WAF}/10^{6})\ \text{U/TB}$$

Other Systems

$$\text{Labor} = (15\text{TF} + 3.3) * (\text{WAF}/10^{6})^{0.5} * \text{XMMAX}^{0.5} * \text{U/TB} * \text{MNTLR}$$

$$\text{Material} = (1.4\text{TF} + 0.8) + 2.3\text{TF} + 0.7) * (\text{TUAFC}/10^{6}) * \text{XMMAX}^{0.5} * \text{U/TB}$$

Structures

$$\text{Labor} = (1.0 + 50 * \text{WAF}/10^{6}) * \text{XMMAX}^{0.5} * \text{U/TB} * \text{MNTLR}$$

$$\text{Material} = (0.3 + 0.8\text{TUAFC}/10^{6}) * \text{XMMAX}^{0.5} * \text{U/TB}$$

Other Power Plant

$$\text{Labor} = (19.0\text{TF} + 0.8) * (\text{WAF}/10^{6})^{0.5} * \text{XMMAX}^{0.5} * \text{U/TB} * \text{MNTLR}$$

$$\text{Material} = 0.3\text{TF} + 0.1 + (0.8\text{TF} + 0.1) * (\text{TUAFC}/10^{6}) * \text{XMMAX}^{0.5}$$

Engine

$$\text{Labor} = [0.4TF + 0.2 + (0.018TF + 0.012) * TCE/XNENGC/10^3]\ U/TB * MNTLR * XNENGC$$

$$\text{Material} = (3.8TF + 2.40)\ (ENGC/10^5)\ U/TB * XNENGC$$

The above formulas calculate the DOC in terms of dollars per aircraft year. This is converted to cents per seat mile by converting the dollars to cents and dividing each element by the seat miles flown per year.

IOC Model

Item I System Expense

System Expense = XKSE x direct maintenance labor dollar

Item II Local Expense

$$\text{Local Expense} = XKLOE \times \frac{\text{maximum takeoff weight}}{1000} \times \text{departures}$$

Item III Aircraft Control

Aircraft Control Expense = XKCO x departures

Item IV Cabin Attendant Expense

Cabin Attendant Expense = XKAT x Cabin attendant block hours

ITEM V Food and Beverage Expense

Food and Beverage Expense = XKFB x weighted revenue passenger block hours

Item VI Passenger Handling Expense

Passenger Handling Expense = XKPH x Passengers enplaned

Item VII Cargo Handling Expense

Cargo Handling Expense = XKCH x Total tons carried

Item VIII Other Passenger Expense -

Other Passenger Expense = XKOP x Revenue Passenger miles

Item IX Other Cargo Expense

Other Cargo Expense = XKOC x Revenue Freight ton miles

Item X General and Administrative Expense

G&A Expense = XKGA x Direct plus indirect Operating expense less depreciation and insurance

DOC and IOC Model Symbol Definitions

XNRITE = Symbol for selecting the range
YEAR = Year input for calculating costs in the proper year's dollars
XNCREW = Number of personnel in the flight crew
XNPASS = Passenger capacity of the aircraft
XNATT = Number of cabin crew
XLF = Load factor
SSFB = Flight crew supersonic flight bonus
IFB = Flight crew international flight bonus
FCSIR = Flight crew salary inflation rate
ENGC = Engine cost per engine
MNTLIR = Maintenance labor inflation rate
TG = Ground time in minutes
U = Utilization
IRA = Insurance rate
PERIOD = Depreciation period
CFT = Cost of fuel ($/LB)
COT = Cost of oil ($/LB)

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MBF = Maintenance burden factor

REVPAS = Number of revenue passengers per year

AVCARG = Average pounds of cargo per flight

XKSE = System expense IOC factor

XKLOE = Local IOC factor

XKCO = Aircraft control IOC factor

XKAT = Cabin attendant IOC factor

XKFB = Food and beverage IOC factor

XKPH = Passenger handling IOC factor

XKCH = Cargo handlingIOC factor

XKOP = Other passenger expense IOC factor

XKOC = Other cargo expense IOC factor

XKGA = G&A expense IOC factor

CFARE = Fare cost factor - (function of distance)

XKFARE = Constant portion of the fare - (function of number of revenue passengers)

XFACC = Facilities cost (dollar input)

TAXR = Income tax rate (decimal)

CARGF = Revenue per cargo ton mile

FB = Block fuel

TB = Block time

WAF = Weight of airframe (weight empty - engines)

TUAFC = Total airframe cost

TF = Flight time

TUACC = Total aircraft flyaway cost including R&D

Return on Investment (ROI) Model

The return on investment (ROI) for the AST is calculated by a simplified method in the ASSET program for comparative analysis.

EVALUATION RESULTS

Five structural approaches were evaluated and compared using the results of the ASSET computer program. The wing design approaches included:

(1) Chordwise stiffened wing arrangement, beaded skin panels, mechanical fasteners (WC1)

(2) Spanwise stiffened wing arrangement, hat stiffened skin panels, mechanical fasteners (WC2)

(3) Monocoque wing arrangement, aluminum brazed honeycomb sandwich skin panels, mechanical fasteners (WC3)

(4) Monocoque wing arrangement, aluminum brazed honeycomb sandwich skin panels, welded (WC4)

(5) Composite reinforced-chordwise stiffened wing arrangement, beaded skin panels, mechanical fasteners (WC5)

In evaluating each of the above wing designs, a skin stringer and frame construction was assumed for the fuselage.

A summary of the ASSET program results are presented in Table 17-2. The results include weight, geometry and cost data for both the constant-size and constant payload-range aircraft.

CONSTANT WEIGHT AIRCRAFT

A comparison of the various parameter (i.e., structural mass, range, cost) for the constant size/weight aircraft indicates a variation in these parameters and the minimum does not necessarily identify the best concepts. The least weight wing

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TABLE 17-2. EVALUATION DATA FOR STRUCTURAL ARRANGEMENTS

STRUCTURAL ARRANGEMENT		CHORDWISE (MECHANICAL) WC1	SPANWISE (MECHANICAL) WC2	MONOCOQUE (MECHANICAL) WC3	MONOCOQUE (WELDED) WC4	CHORDWISE (COMP. REINF.) WC5
CONSTANT GTOW:						
TOGW	(LB)	750000	750000	750000	750000	750000
OWE	(LB)	329474	328315	315982	318759	313125
WING WEIGHT	(LB)	106010	104834	92330	95146	89434
WING AREA	(FT^2)	10822	10822	10822	10822	10822
WING UNIT WT	(LB.FT^{-2})	9.80	9.69	8.53	8.79	8.26
RANGE	(N·Mi)	3830	3870	4123	4066	4166
FLYAWAY COST	(MIL DOL)	90.65	89.19	104.93	104.79	93.81
DOC	(C/SM)	1.94	1.92	2.03	2.04	1.92
IOC	(C/SM)	0.94	0.93	0.91	0.91	0.90
ROI A.T.	(%)	1.12	1.48	0.30	0.06	1.74
CONSTANT PAYLOAD-RANGE:						
TOGW	(LB)	884847	867126	772641	789992	759498
OWE	(LB)	381691	373353	324109	333338	316481
WING WEIGHT	(LB)	129895	125254	95682	101296	90785
WING AREA	(FT^2)	12768	12512	11149	11399	10959
WING UNIT WT	(LB.FT^{-2})	10.20	10.00	8.58	8.89	8.28
RANGE	(N·Mi)	4200	4200	4200	4200	4200
FLYAWAY COST	(MIL DOL)	103.19	99.83	107.52	109.39	94.73
DOC	(C/SM)	2.14	2.09	2.06	2.11	1.93
IOC	(C/SM)	0.94	0.94	0.91	0.91	0.90
ROI A.T.	(%)	–1.37	–0.74	–0.11	–0.64	1.55

concept is the composite reinforced design (WC5); the spanwise design (WC2) is, however, the least initial cost concepts as typified by the flyaway cost. It can be seen from the tabulated data that the weight savings realized by the application of composites to the spar caps permits approximately 16,600 pounds of additional fuel to be carried. Hence, the range capability of the structurally efficient composite reinforced design (WC5) is approximately 340 nautical miles greater than the chordwise wing design (WC1). All concepts, however, do not meet the range criteria of 4200 nautical miles. A reduction in structural mass of 1700 pounds is required to the composite reinforced design to satisfy the payload-range requirement.

CONSTANT PAYLOAD-RANGE AIRCRAFT

The constant payload-range data of Table 17-2 indicates that the takeoff gross weight of the resized aircraft varies from a maximum of 885,000 pounds to a minimum of 760,000 pounds. The data presents the minimum structural weight, size and cost to be the composite reinforced design (WC5).

Growth Factor. - Figure 17-5 visually displays the takeoff gross weight variation with the range capability of the constant-weight and constant payload-range aircraft. The trends presented indicates that the growth factor of this class of aircraft, which is represented by approximately 50-percent fuel, is 6 (i.e., a 1-pound increase in structural weight results in a 6-pound increase in the aircraft takeoff gross weight). Thus, the importance of minimizing structural mass is emphasized by these trends.

Structural Mass Versus Cost. - The wing mass variation with relative cost is presented in Figure 17-6. The data relates the structural efficiency of the design concepts with the direct operating cost, normalized to the least cost approach. DOC is a measure of total-system-cost that includes such parameters as: structural efficiency; engine requirement variation as the airplane size changes; the impact of airplane size, concept selection, and mass on fuel consumption; and development, manufacturing and operating (maintenance) costs. The desirable area, as visually displayed by the arrow, is the least weight and least cost region. The composite reinforced design (WC5) is the least weight and cost concept evaluated.

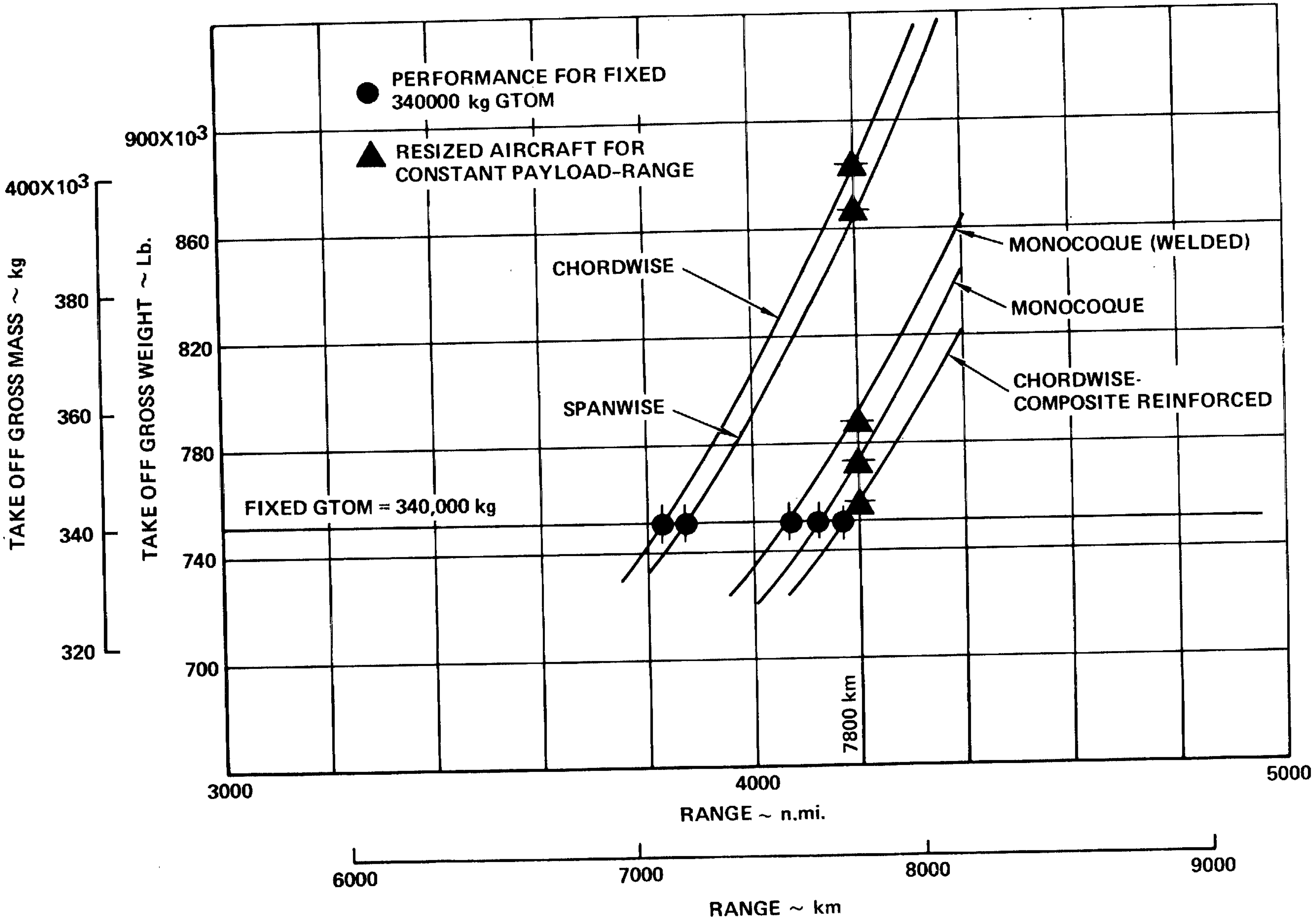

FIGURE 17-5. AIRCRAFT SIZING RESULTS

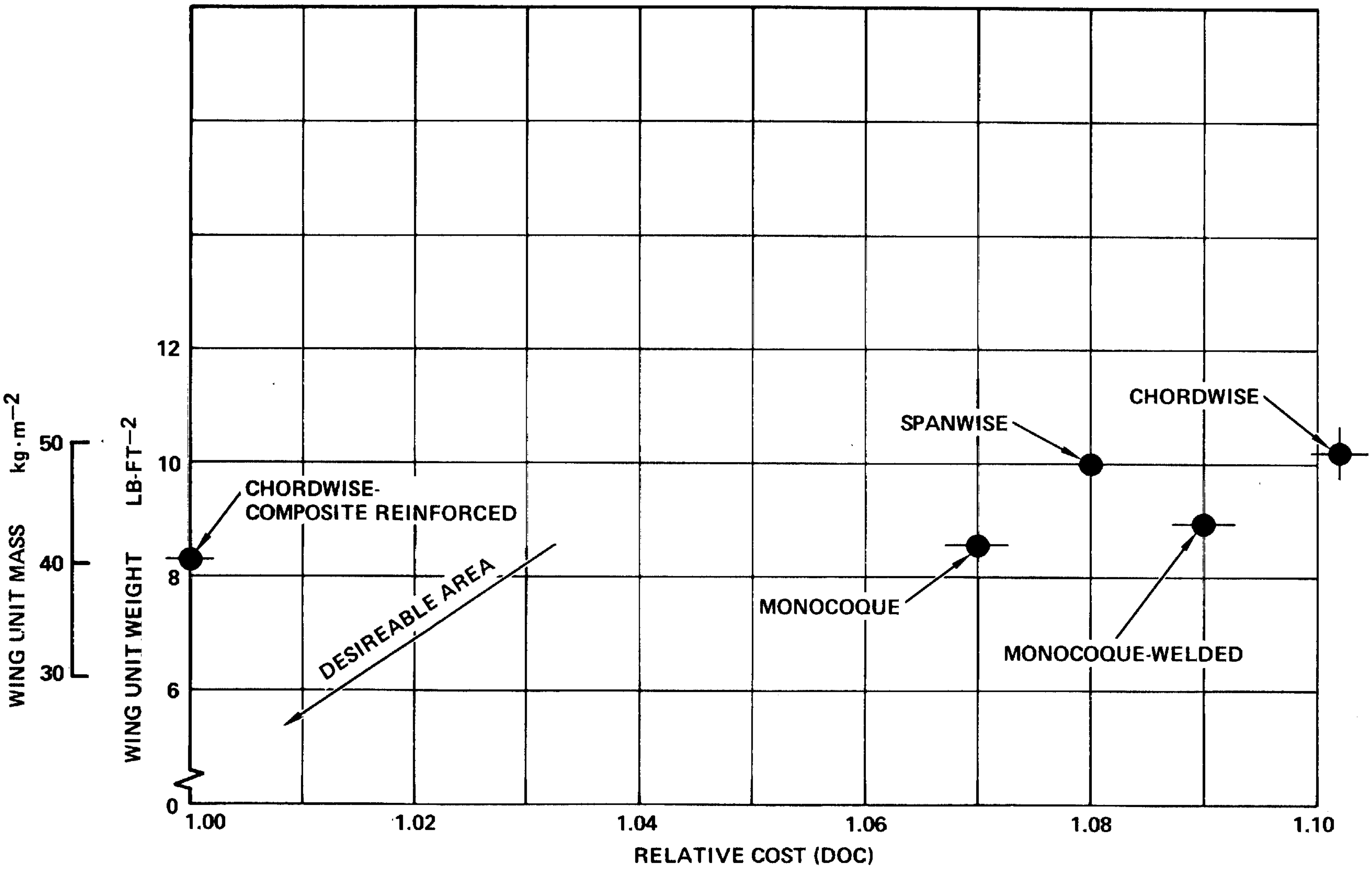

FIGURE 17-6. STRUCTURAL MASS VERSUS COST

Technology Improvement Versus Cost. - Figure 17-7 presents a bar chart relating the technology level of each design concept with the initial cost. The metallic stiffened skin approaches are displayed on the left hand side with the more advanced technology concepts on the right hand side. The initial cost is measured by the flyaway cost of the vehicles, normalized to the least cost system. The composite reinforced design (WC5) is the least cost with the monocoque approaches being approximately 15-percent greater.

Technology Risk Versus Cost. - Technology risk, as presented herein, is a quantitative factor related to the damage tolerance characteristics of the design concepts, tempered with an assessment of development risk. As shown in Figure 17-8 the spanwise and chordwise designs are considered as the least risk concepts; with the welded monocoque design representing the highest risk. Since the composite reinforced design is essentially a chordwise stiffened design with only the spar caps reinforced with Boron Polyimide (B/PI) composites, it is displayed above the chordwise design. The monocoque design is represented as nominal. The cost parameters are normalized (to the least cost concept) direct operating costs. The composite reinforced design is the most desirable of the five design concepts evaluated.

Investment Versus Cost. - Figure 17-9 presents the investment cost variation for each design concept with direct operating costs. Both cost values are normalized to the least cost approach.

Investment cost includes the cost for the aircraft, the aircraft spares and the special support equipment. The cost of the production aircraft is the primary element of investment. The spares and support equipment are treated as percentages of the flyaway cost of the aircraft.

As shown by the normalized values, the composite reinforced design is the least cost approach.

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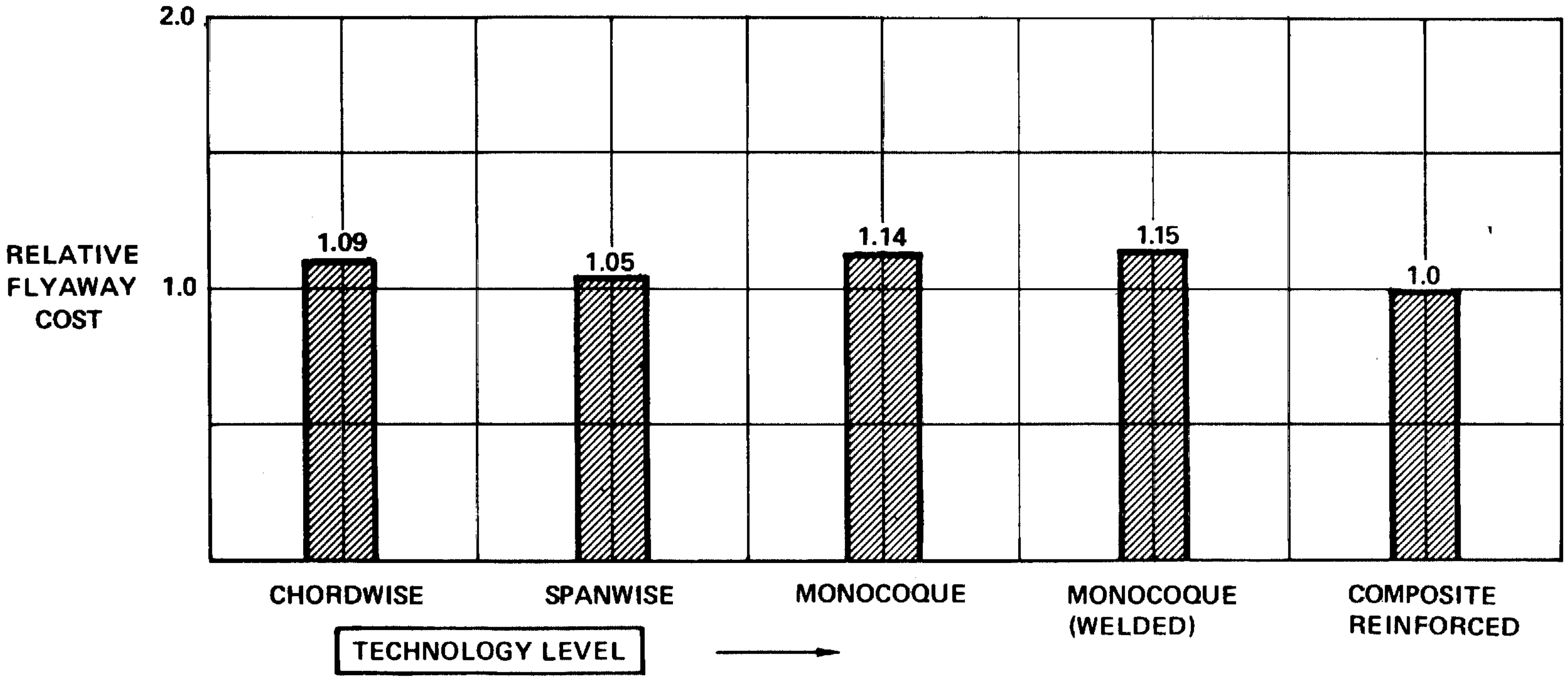

FIGURE 17-7. TECHNOLOGY IMPROVEMENT VERSUS COST

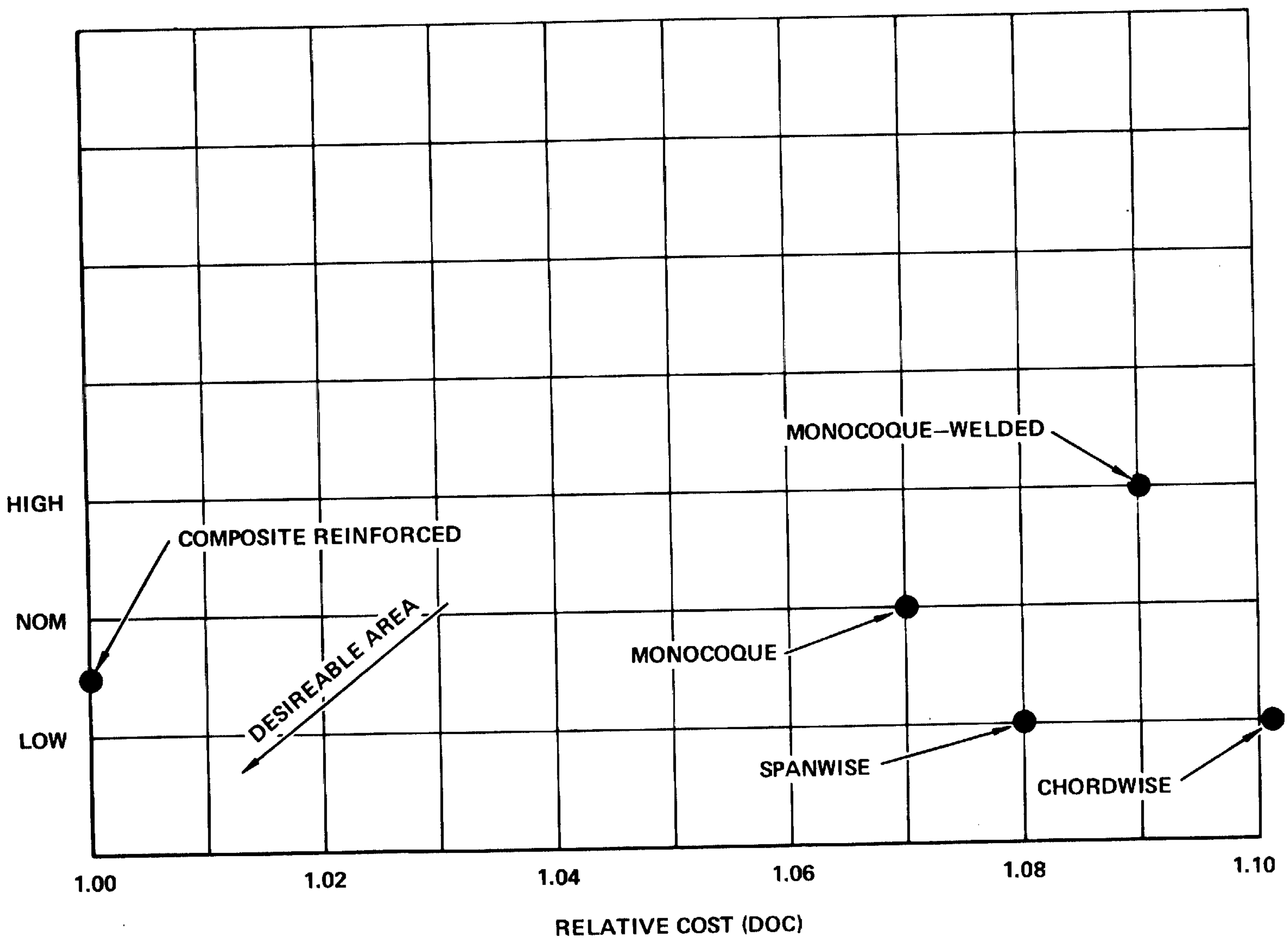

FIGURE 17-8. TECHNICAL RISK VERSUS COST

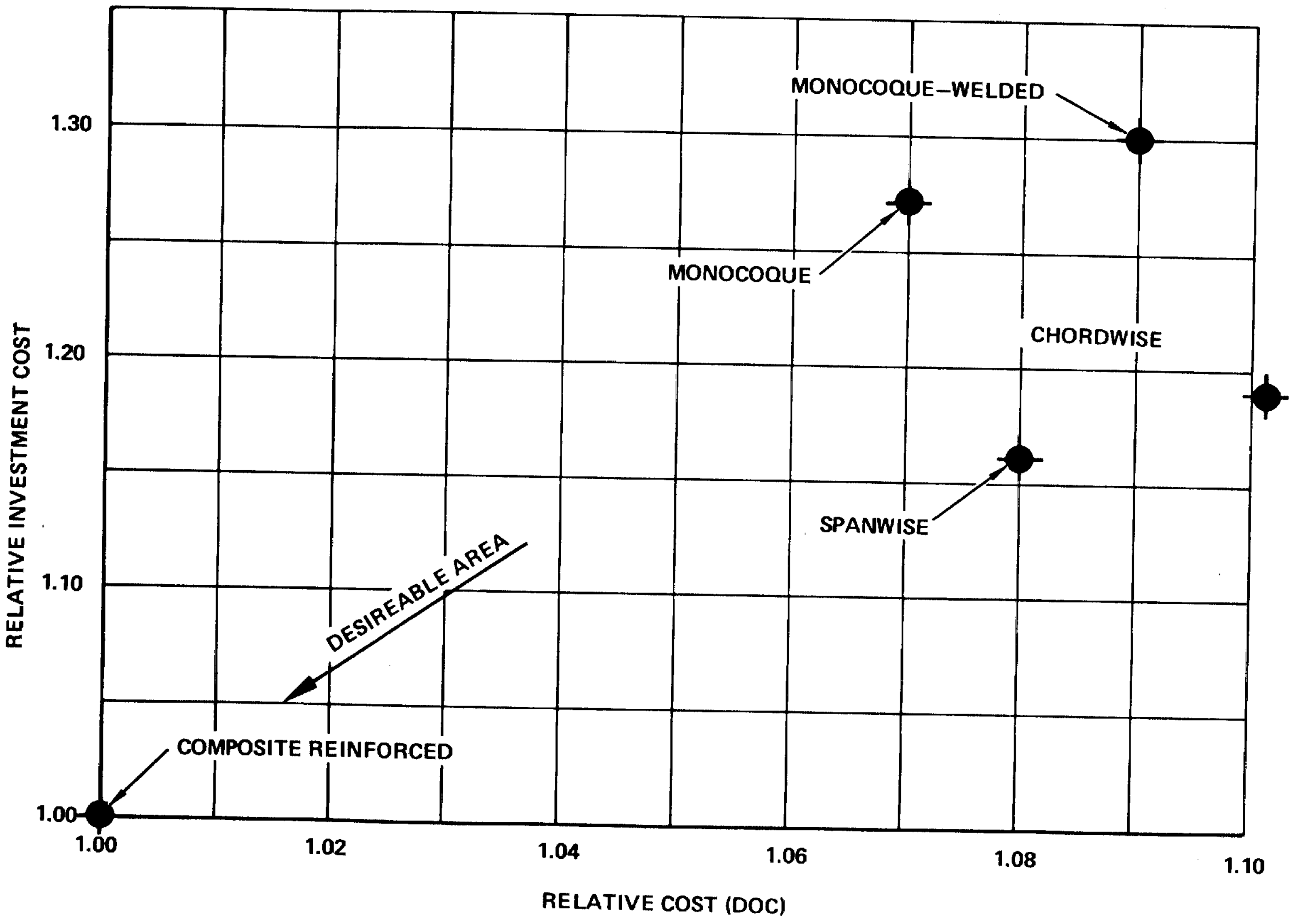

FIGURE 17-9. INVESTMENT VERSUS COST

CONCEPT SELECTION

The various wing design concepts, each with a skin-stringer fuselage design, were evaluated with respect to structural mass, performance and cost. These factors were interrelated to yield a relative comparison, based on minimum-total-system-cost for a constant payload-range aircraft.

Based on a constant-weight airplane the ranking of the design concepts shown in Table 17-3 was obtained. When these design concepts were applied to a minimum total-system-cost airplane the ranking of the concepts was unchanged (Table 17-4). A comparison of the relative costs of the various wing design concepts is also presented in Table 17-4, and shows that the composite reinforced design is 7 to 11 percent less costly than the other design concepts.

The best homogeneous (single concept applied to total wing) structural approach for design of Mach 2.7 supersonic cruise aircraft is the least cost and weight chordwise stiffened design with metallic surface panels and composite reinforced spars. The structural arrangement for this design concept is presented in Figure 17-10. Approximately 6000 pounds of composite material is used and results in a 16,600 pound weight saving of wing structural mass. Structurally efficient circular arc-convex beaded surface panels of titanium alloy (Ti-6Al-4V annealed) are used. The surface panel design is directed towards alleviating thermal stresses while utilizing both skins in resisting shear. With the beaded inner and outer skins, the surfaces do not participate in resisting wing bending and all the bending material is concentrated in the spar caps. The titanium spar caps are reinforced with boron/polyimide as shown in the figure. The reinforcement strips are continuous from BL 470L to BL 470R.

The importance of minimum mass structural concepts was emphasized by the increasing cost trends with an increase in wing structural mass as shown in Table 17-4. Weight inefficiencies evaluated under the payload-range constraints can and do raise costs appreciably. Furthermore, with the apparent high growth factor (GF ≈ 6) of this class of aircraft, it appears that considerable effort is warranted to remove unnecessary weight to minimize the cascading effect on aircraft size and takeoff gross weight.

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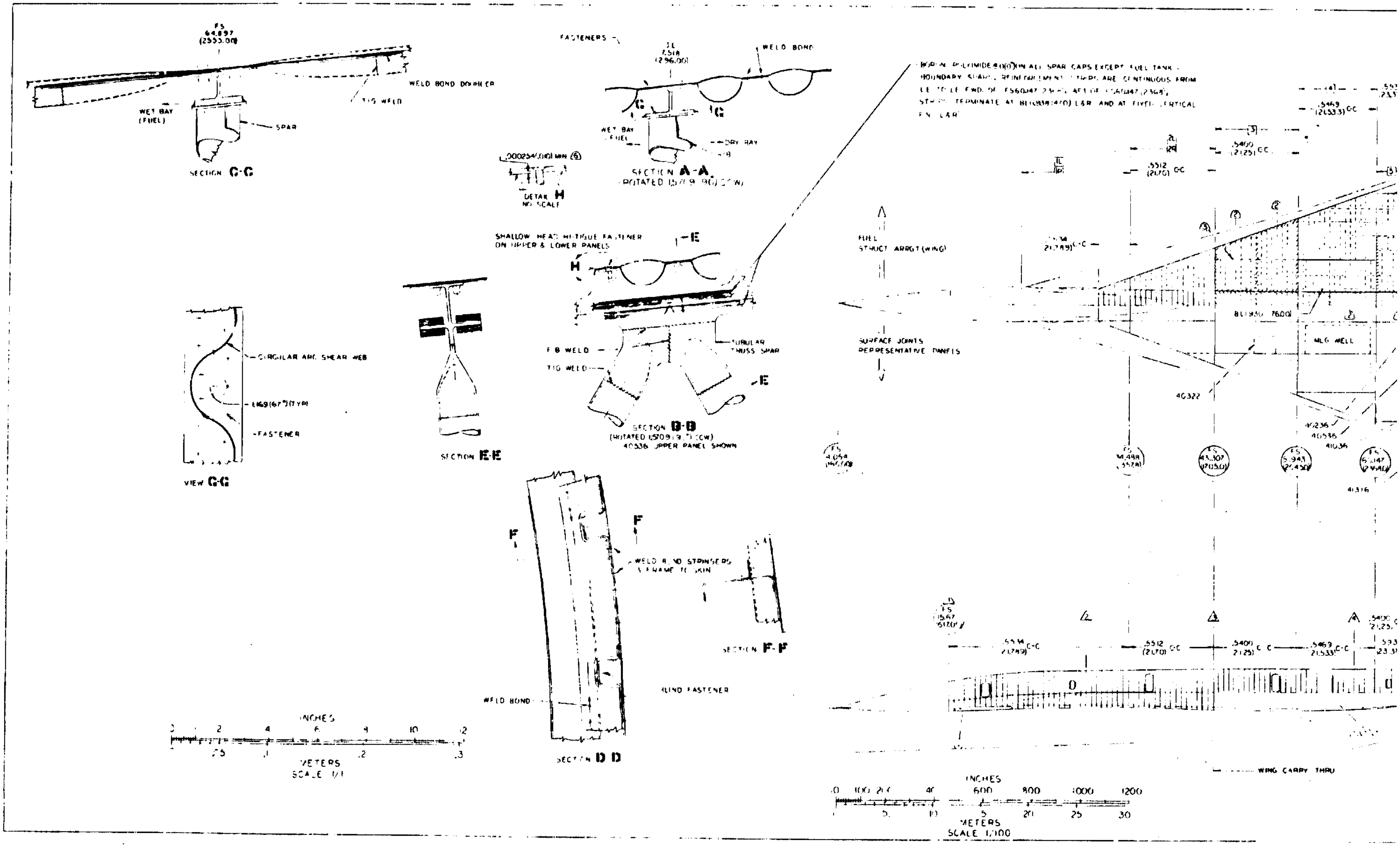
TABLE 17-3. CONCEPT EVALUATION SUMMARY - CONSTANT WEIGHT AIRCRAFT

CONCEPT	WING WEIGHT LBS/FT^2	RELATIVE WEIGHT
(1) CHORDWISE STIFFENED – CONVEX-BEADED PANELS	9.80	1.19
(2) SPANWISE STIFFENED – HAT-STIFFENED PANELS	9.68	1.17
(3) MONOCOQUE – ALUMINUM BRAZED HONEYCOMB CORE PANELS	8.54	1.03
(4) MONOCOQUE – ALUMINUM BRAZED HONEYCOMB CORE PANELS (WELDED)	8.85	1.07
(5) CHORDWISE STIFFENED – CONVEX-BEADED PANELS; B/PI REINFORCED SPARS	8.25	1.00

TABLE 17-4. CONCEPT EVALUATION SUMMARY - CONSTANT PAYLOAD-RANGE AIRCRAFT

STRUCTURAL ARRANGEMENT AND CONCEPT	WING MASS			COST	
	kg • m^{-2}	LB • FT^{-2}	RELATIVE MASS	DOC (C/SM)	RELATIVE COST
(1) CHORDWISE STIFFENED – CONVEX-BEADED PANELS	49.80	10.20	1.23	2.14	1.11
(2) SPANWISE STIFFENED – HAT-STIFFENED PANELS	48.82	10.00	1.21	2.09	1.08
(3) MONOCOQUE – ALUMINUM BRAZED HONEYCOMB CORE PANELS	41.89	8.58	1.04	2.06	1.07
(4) MONOCOQUE – ALUMINUM BRAZED HONEYCOMB CORE PANELS (WELDED)	43.40	8.89	1.07	2.11	1.09
(5) CHORDWISE STIFFENED – CONVEX-BEADED PANELS; B/PI REINFORCED SPARS	40.43	8.28	1.00	1.93	1.00


FS 64.897 (2555.01)
WELD BOND DOUBLER
TIG WELD
WET BAY (FUEL)
SPAR
SECTION G-G
FASTENERS
BL 7.518 (296.00)
WELD BOND
G
G
WET BAY FUEL
DRY BAY
SECTION A-A
(ROTATED 1.571(9.90) CW)
.000254(.010) MIN (6)
DETAIL H
NO SCALE
SHALLOW HEAD HI-TIGUE FASTENER ON UPPER & LOWER PANELS
H
E
E
TUBULAR TRUSS SPAR
E.B WELD
TIG WELD
SECTION B-B
(ROTATED 1.5709(9.0) CW)
40536 UPPER PANEL SHOWN
CIRCULAR ARC SHEAR WEB
1.169(67°)(TYP)
FASTENER
VIEW C-C
SECTION E-E
F
F
WELD BOND STRINGERS & FRAME TO SKIN
SECTION F-F
BLIND FASTENER
WELD BOND
SECTION D-D
INCHES
0 2 4 6 8 10 12
.05 .1 .2 .3
METERS
SCALE 1/1
BORON POLYIMIDE (10) IN ALL SPAR CAPS EXCEPT FUEL TANK BOUNDARY SPARS. REINFORCEMENT STRIPS ARE CONTINUOUS FROM LE TO LE END OF FS60.047 (2364), AFT OF FS60.047 (2364), STRIPS TERMINATE AT BL(938)(470) L&R AND AT FIXED VERTICAL FIN L&R
FUEL
STRUCT ARRGT (WING)
SURFACE JOINTS REPRESENTATIVE PANELS
.5534 (21.789) C-C
.5512 (21.70) C-C
.5400 (21.25) C-C
.5469 (21.533) C-C
BL 1.930 (76.00)
MLG WELL
40322
40236
40536
41036
41316
FS 41.054 (1617.00)
FS 44.498 (1752.8)
FS 43.307 (1705.0)
FS 5.943 (2.450)
FS 6.147 (2.420)
FS 1.567 (51.20)
.5400 (21.25) C-C
.5463 (21.533) C-C
.553 (23.3)
WING CARRY THRU
INCHES
0 100 200 400 600 800 1000 1200
5 10 15 20 25 30
METERS
SCALE 1/100

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FOLDOUT FRAME 2

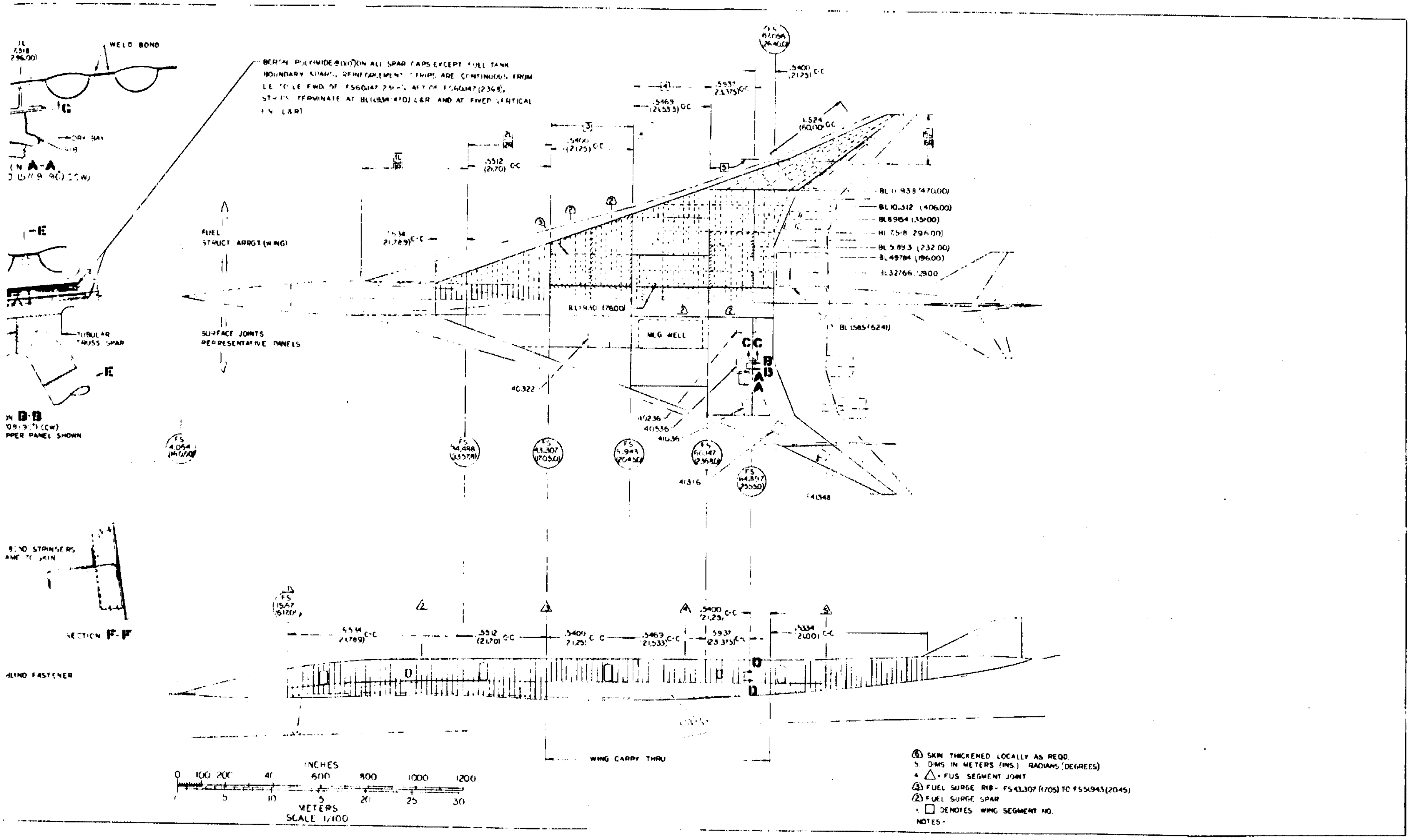
WELD BOND
JL 7.518 (296.00)
C
DRY BAY
N A-A
FUEL STRUCT ARRGT (WING)
SURFACE JOINTS REPRESENTATIVE PANELS
-E
TUBULAR TRUSS SPAR
N B-B
PPER PANEL SHOWN
BORON POLYIMIDE (0/0) ON ALL SPAR CAPS EXCEPT FUEL TANK BOUNDARY SPARS. REINFORCEMENT STRIPS ARE CONTINUOUS FROM LE TO LE FWD OF FS60.147 (2368), AFT OF FS60.147 (2368), STRIPS TERMINATE AT BL(11.938 (470) L&R AND AT FIXED VERTICAL FN (L&R)
.5512 (21.70) C-C
.5400 (21.25) C-C
.5469 (21.533) C-C
.5937 (23.375) C-C
.5400 (21.25) C-C
1.524 (60.00) C-C
.5514 (21.789) C-C
.5334 (21.00) C-C
FS 67.056 (2640.0)
FS 4.064 (160.00)
FS 34.488 (1357.8)
FS 43.307 (1705.0)
FS 51.943 (2045.0)
FS 60.147 (2368.0)
FS 64.897 (2555.0)
FS 15.67 (617.0)
BL 11.938 (470.00)
BL 10.312 (406.00)
BL 8.954 (351.00)
BL 7.518 (296.00)
BL 5.893 (232.00)
BL 4.9784 (196.00)
BL 3.2766 (129.00)
BL 1.585 (62.41)
BL 1.930 (76.00)
MLG WELL
C-C
B-B
A-A
D-D
40322
40236
40536
41036
41316
41348
BOND STRINGERS SAME TO SKIN
SECTION F-F
BLIND FASTENER
WING CARRY THRU
INCHES
0 100 200 400 600 800 1000 1200
METERS
0 5 10 15 20 25 30
SCALE 1/100
NOTES-
1. □ DENOTES WING SEGMENT NO.
② FUEL SURGE SPAR
③ FUEL SURGE RIB - FS43.307 (1705) TO FS51.943 (2045)
4. △ = FUS SEGMENT JOINT
5. DIMS IN METERS (INS.) RADIANS (DEGREES)
⑥ SKIN THICKENED LOCALLY AS REQD.

FIGURE 17-10. COMPOSITE REINFORCED WING DESIGN - FASTENERS

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FOLDOUT FRAME 3

The structural weight for the various regions of the wing structure are shown in Table 17-5 for the five structural approaches. It appears that incorporating the minimum weight regions into the wing design would result in the best approach for a Mach 2.7 design.

Thus, the recommended structural approach for the Task II Detailed Engineering Design and Analysis is a hybrid structural approach consisting of the monocoque and chordwise stiffened structural arrangement as shown in Figure 17-11. A minimum structural mass airplane (near-term) is obtained by combining the minimum mass components (i.e., wing forward box, aft box and tip) as determined by the detailed structural analysis. Table 17-6 shows the airplane weight and cost parameters for this hybrid design, for both constant-weight and constant payload-range criteria. The hybrid design very nearly satisfies the payload-range requirement specified for the 750,000 pound baseline configuration concept.

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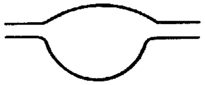
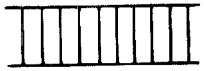
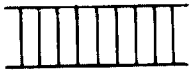
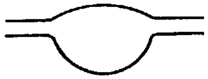
TABLE 17-5. WING WEIGHTS FOR STRUCTURAL ARRANGEMENTS

WING WEIGHT AND SEGMENT	STRUCTURAL ARRANGEMENT				
	CHORDWISE	SPANWISE	MONOCOQUE	MONOCOQUE	CHORDWISE
	WELD BOND  MECH. FASTEN.	WELD BOND MECH. FASTEN.	ALUM BRAZED  MECH. FASTEN.	ALUM BRAZED  WELDED	COMP. REINF.  SPARS ONLY
VARIABLE WEIGHT	64,658	63,482	50,978	53,794	48,082
● FWD. BOX	(22,090)	(25,364)	(21,982)	(24,057)	(20,580)
● AFT BOX	(29,016)	(25,242)	(19,692)	(20,153)	(17,384)
● TIP	(13,552)	(12,876)	(9,304)	(9,584)	(10,118)
FIXED WEIGHT	41,352	41,352	41,352	41,352	41,352
Σ TOTAL~LB	106,010	104,834	92,330	95,146	89,434

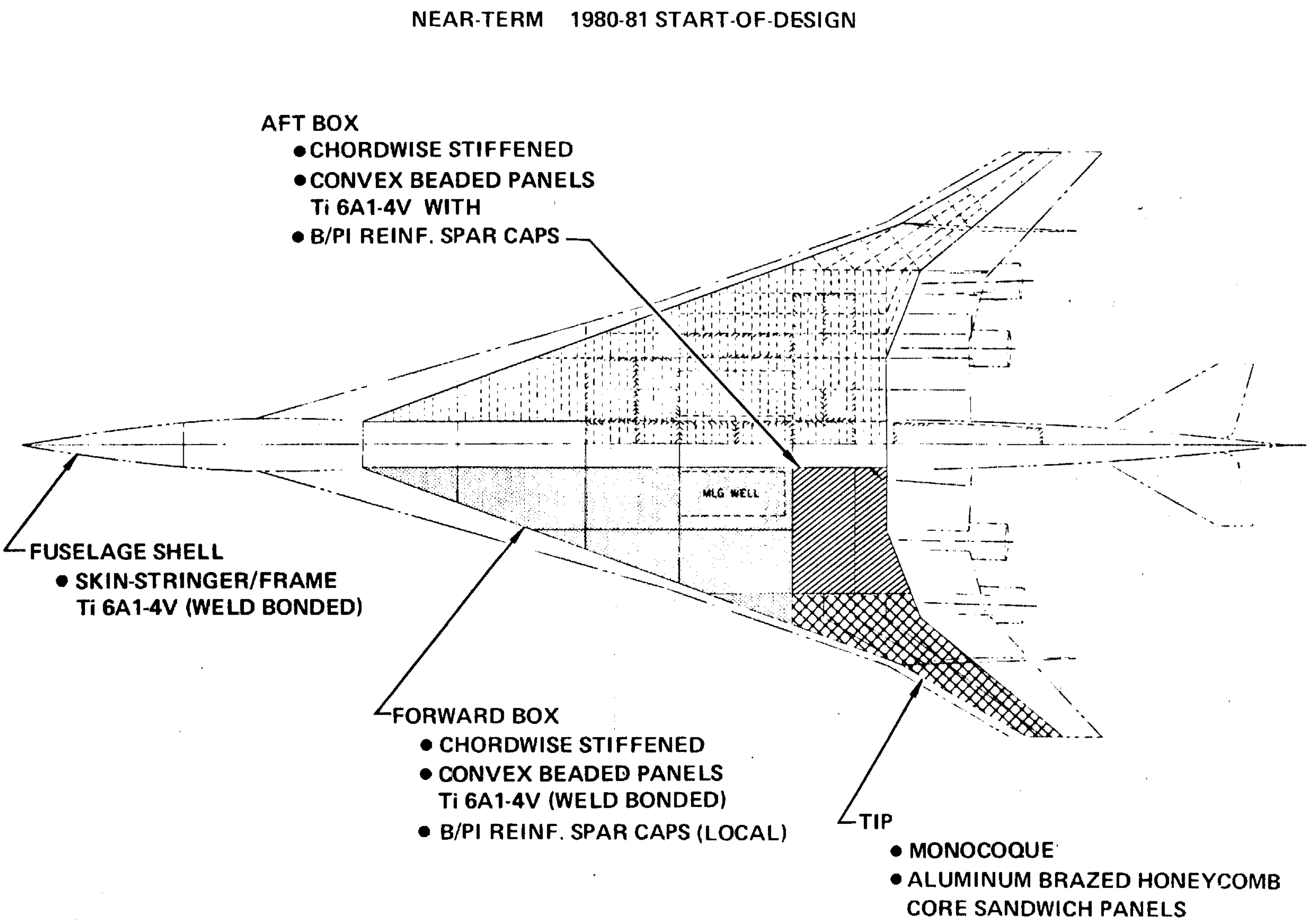

NEAR-TERM 1980-81 START-OF-DESIGN
AFT BOX
● CHORDWISE STIFFENED
● CONVEX BEADED PANELS
Ti 6A1-4V WITH
● B/PI REINF. SPAR CAPS
MLG WELL
FUSELAGE SHELL
● SKIN-STRINGER/FRAME
Ti 6A1-4V (WELD BONDED)
FORWARD BOX
● CHORDWISE STIFFENED
● CONVEX BEADED PANELS
Ti 6A1-4V (WELD BONDED)
● B/PI REINF. SPAR CAPS (LOCAL)
TIP
● MONOCOQUE
● ALUMINUM BRAZED HONEYCOMB
CORE SANDWICH PANELS

FIGURE 17-11. STRUCTURAL APPROACH FOR TASK II

TABLE 17-6. EVALUATION DATA FOR HYBRID STRUCTURAL ARRANGEMENT

STRUCTURAL ARRANGEMENT		HYBRID (MECHANICAL) WC7
CONSTANT GTOW:		
TOGW	(LB)	750000
OWE	(LB)	312322
WING WEIGHT	(LB)	88620
WING AREA	(FT^2)	10822
WING UNIT WT	($LB.FT^{-2}$)	8.19
RANGE	(N.Mi)	4183
FLYAWAY COST	(MIL DOL)	93.57
DOC	(C/SM)	1.91
IOC	(C/SM)	0.90
ROI A.T.	(%)	1.82
CONSTANT PAYLOAD-RANGE:		
TOGW	(LB)	754665
OWE	(LB)	313963
WING WEIGHT	(LB)	89216
WING AREA	(FT^2)	10889
WING UNIT WT	($LB.FT^{-2}$)	8.20
RANGE	(N.Mi)	4200
FLYAWAY COST	(MIL DOL)	94.02
DOC	(C/SM)	1.92
IOC	(C/SM)	0.90
ROI A.T.	(%)	1.73

C13

REFERENCES

1 Levenson, G.S.; Boren, H.E.; Tihansky, D.P.; Timson, F.S., "Cost-Estimating Relationships for Aircraft Airframes" R-761-PR (abridged), Rand, Santa Monica, February 1972.

2 Large, J.P., "Estimating Aircraft Turbine Engine Costs" RM-6384/1-PR, Rand, Santa Monica, September 1970.

3 Standard Method of Estimating Comparative Direct Operating Costs of Turbine Powered Transport Airplanes, Air Transport Association, December 1967.

4 "More Realism in a Standard Method for Estimating Airline Operating Expense," Lockheed California Company, Report OEA/SST/179, March 1966.

5 "Revisions to 1964 Lockheed/Boeing Indirect Operating Expense Method," Lockheed California Company, Report COA2061, December 1969.

6 Indirect Operating Expense Coefficients years 1963 through 1972, Lockheed California Company, COA/1277, June 1973.

APPENDIX A

COMPUTER PRINTOUT

ASSET PARAMETRIC ANALYSIS

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T/C	AR	W/S	T/W
2.65	1.62	69.3	0.477

W E I G H T   S T A T E M E N T

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
TAKE-OFF WEIGHT	(759496.)		
FUEL AVAILABLE	394017.	FUEL	51.88
ZERO FUEL WEIGHT	(365461.)		
PAYLOAD	49000.	PAYLOAD	6.45
OPERATING WEIGHT	(316461.)		
OPERATING ITEMS	5476.	OPERATING ITEMS	1.43
STANDARD ITEMS	5365.		
EMPTY WEIGHT	(305620.)		
WING	90765.		
TAIL	11545.		
BODY	42959.	STRUCTURE	25.04
LANDING GEAR	30706.		
SURFACE CONTROLS	8517.		
NACELLE AND ENGINE SECTION	5688.		
PROPULSION	(80189.)	PROPULSION	10.56
WEIGHT OF LIFT ENGINES	0.		
VECTOR CONTROL SYSTEM	0.		
ENGINES	51772.		
THRUST REVERSAL	0.		
AIR INDUCTION SYSTEM	21050.		
FUEL SYSTEM	5729.		
ENGINE CONTROLS + STARTER	1639.		
INSTRUMENTS	1239.		
HYDRAULICS	5772.		
ELECTRICAL	4557.	EQUIPMENT	4.64
AVIONICS	1900.		
FURNISHINGS AND EQUIPMENT	11500.		
ENVIRONMENTAL CONTROL SYSTEM	6300.		
AUXILIARY GEAR	1980.		
A.M.P.R.	(236479.)	TOTAL	(100.00)
EXCESS FUEL CAPACITY - BODY	837998.		
EXCESS FUEL CAPACITY - WING	0.		
EXCESS BODY LENGTH - FT	0.0		
STRUCTURE ALUMINUM			

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                                         W E I G H T S   M A T R I X

             / MATERIAL
  ELEMENT/                                    AL       TI.     STEEL     COMP.     OTHER     TOTAL

                              WING          4178.    77712.    1816.    5629.     1453.    90788.

                              TAIL           520.    10726.     115.       0.      185.    11545.
  ------------------------------------------------------------------------------------------------
                              FUSEL         1975.    34093.     773.    1073.     5024.    42939.

                              L. G.           31.     7677.   11792.       0.    11209.    30708.

                              NACELLE        108.      845.    1891.       0.        0.     2844.
  ------------------------------------------------------------------------------------------------
                              AIR INDUCT     966.    18650.     210.       0.     1221.    21050.

                              S. CTLS       2044.      383.    1789.     126.     4173.     8517.

                              TOTALS        9822.   150086.   18386.    6830.    23264.   208388.

                                   C O N F I G U R A T I O N   G E O M E T R Y

                  BASIC WING--     AREA(SQ.FT)    SPAN(FT)  TAPER RATIO  C/4 SWEEP  L.E. SWEEP  CR(FT)       MAC(FT)
                                     10959.1       133.34       0.0       66.626      72.500     164.38       109.59

                  INBOARD WING --  AREA(SQ.FT)    EXP. AREA L.E. SWEEP   REF L(FT)  SFLE(SQ.FT) AVG T/C
                                     10959.1       9457.3      72.50      101.81       0.0         2.65

                  OUTBOARD WING-- AREA(SQ.FT)    Y BRK(FT) L.E. SWEEP   REF L(FT)  SFLE(SQ.FT) AVG T/C
                                       0.0           0.0      72.50      101.81       0.0         2.65

                  TOTAL WING--     AREA(SQ.FT)     EFF AR    AVG T/C      CR(FT)      CT(FT)    (B/2)/LW        P
                                     10959.1         1.62       2.65      164.38       0.0        0.315       0.389

                  WING TANK--        CBAR1(FT)   CBAR2(FT)     FTL(FT)  FVWING(CU FT)  FVBOX(CU FT)
                                      150.41        0.0        61.00     4576.30     2601.62

                  FUSELAGE--       LENGTH(FT)   S WET(SQ FT)   BWW(FT)   EQUIV D(FT)  SPI(SQ FT)
                                      297.00      7876.0         9.47      13.22       137.30

                                       BW(FT)       BH(FT)    SBW(SQ FT)   FVL(CU FT)
                                       11.33        11.50      9177.00    20000.00

                  TAIL--           SHT(SQ.FT)   SHTX(SQ.FT) HT REF L(FT) SVT(SQ.FT)   SVTX(SQ.FT)  VT REF L(FT)
                                     1106.75       934.04      23.64       471.90       471.90      26.02

                  PROPULSION--       ENG L(FT)    ENG D(FT)    POD L(FT)    POD D(FT)    POD S WET     NO. PODS  INLET L(FT)
                                       22.39         7.13       39.39         8.06      4001.53          4.         0.0
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                                     M I S S I O N   S U M M A R Y

          INTERNATIONAL MISSION  STD DAY + 14.4 F
SEGMENT       INIT     INIT    INIT     SEGMT    TOTAL   SEGMT    TOTAL   SEGMT   TOTAL   EXTERN   ENGINE   EXTERN    AVG     AVG     MAX
            ALTITUDE   MACH   WEIGHT    FUEL     FUEL    DIST     DIST    TIME    TIME    STORE    THRUST   F TANK    L/D     SFC     OVER
              (FT)      NO     (LB)     (LB)     (LB)   (N MI)  (N MI)   (MIN)   (MIN)   TAB ID   TAB ID   TAB ID   RATIO  (FF/T)   PRES

 TAKEOFF
   POWER 1       0.   0.0    759496.    3614.    3614.      0.      0.    10.0    10.0       0.  -97101.       0.    0.0    0.571   0.0

   POWER 2       0.   0.300  755885.    6112.    9725.      0.      0.     0.5    10.5       0.   97201.       0.    5.18   1.388   0.0

 CLIMB           0.   0.300  749773.    7262.   16987.      4.      4.     1.1    11.5       0.   97201.       0.    7.45   1.446   0.0

 CRUISE       5000.   0.413  742511.    4352.   21339.      0.      4.     4.0    15.5       0.  -97101.       0.    8.24   0.727   0.0

 ACCEL        5000.   0.413  738159.    1793.   23132.      1.      6.     0.3    15.8       0.   97201.       0.    9.96   1.472   0.0

 CLIMB        5000.   0.537  736366.   22784.   45916.     36.     42.     4.7    20.5       0.   97201.       0.   11.26   1.508   0.0

 CLIMB       34000.   0.964  713582.   49852.   95769.    254.    296.    14.2    34.7       0.   97201.       0.    8.34   1.533   0.0

 CLIMB       63000.   2.605  663729.       0.   95769.      0.    296.     0.0    34.7       0.   97201.       0.    8.74   1.563   0.0

 CRUISE      63000.   2.620  663729.  227210.  322979.   3694.   3990.   144.7   179.5       0.  -97201.       0.    8.63   1.498   0.0

 DECEL       73000.   2.620  436519.      97.  323076.     23.   4013.     0.9   180.4       0.   97501.       0.    8.44  -0.769   0.0

 DESCENT     73000.   2.369  436422.    1300.  324375.    154.   4167.    13.1   193.5       0.   97501.       0.    9.17  -0.386   0.0

 CRUISE      73000.   2.620  435123.    1679.  326054.     33.   4200.     1.3   194.8       0.  -97201.       0.    8.50   1.511   0.0

 CRUISE       5000.   0.413  433444.    2667.  328721.      0.   4200.     5.0   199.8       0.  -97101.       0.   11.18   0.830   0.0

 RESET           0.   0.0    430777.       0.  328721.      0.   4200.     0.0   199.8       0.       0.       0.    0.0    0.0     0.0

 RESET           0.   0.0    430777.       0.  328721.  -4200.      0.   *****     0.0       0.       0.       0.    0.0    0.0     0.0

 RESERVE         0.   0.0    430777.   23010.  351731.      0.      0.     0.0     0.0       0.       0.       0.    0.0    0.0     0.0

 CLIMB           0.   0.200  407767.    2774.  354505.      2.      2.     0.4     0.4       0.   97201.       0.    9.73   1.444   0.0

 CLIMB        1500.   0.505  404993.   13428.  367933.     22.     23.     2.9     3.3       0.   97201.       0.    9.64   1.503   0.0

 CRUISE      43000.   0.900  391565.   10500.  378433.    167.    190.    19.0    22.4       0.  -97101.       0.   10.94   0.937   0.0

 DESCENT     43000.   0.900  381065.     783.  379215.     56.    246.     7.9    30.3       0.   97501.       0.    9.89  -0.491   0.0

 CRUISE      43000.   0.900  380282.     769.  379984.     12.    260.     1.4    31.7       0.  -97101.       0.   10.93   0.938   0.0

 LOITER      15000.   0.509  379513.   14031.  394015.      0.    260.    30.0    61.7       0.  -97101.       0.   11.30   0.851   0.0

 TOGWT=   759496.3      FUEL A=394017.2      FUEL R=394014.9
```

PRODUCTION

	PRODUCTION YEARS										
	1	2	3	4	5	6	7	8	9	10	TOTAL
AIRFRAME	1242.72	1170.85	1280.96	1406.62	1528.95	1417.99	1339.83	1280.41	1232.97	1193.80	13065.29
ENGINEERING											
HOURS	4416.	3804.	4025.	4276.	4521.	4097.	3801.	3577.	3399.	3252.	39171.
LABOR RATE	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	
OVERHEAD RATE	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	
TOTAL	76.71	66.08	69.92	74.30	78.53	71.17	66.03	62.13	59.04	56.49	680.40
TOOLING											
HOURS	5300.	4565.	4830.	5133.	5425.	4917.	4561.	4292.	4079.	3903.	47005.
LABOR RATE	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	
OVERHEAD RATE	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	
TOTAL	97.78	84.22	89.12	94.71	100.09	90.72	84.16	79.19	75.25	72.01	867.25
MANUFACTURING											
HOURS	44163.	38641.	40254.	42776.	45209.	40975.	38011.	35770.	33989.	32524.	391711.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	699.55	612.57	637.62	677.57	716.11	649.04	602.10	566.59	538.38	515.16	6204.70
QUALITY CONTROL											
HOURS	8833.	7606.	8051.	8555.	9042.	8195.	7602.	7154.	6798.	6505.	78342.
LABOR RATE	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	150.24	129.41	136.94	145.52	153.80	139.40	129.31	121.69	115.63	110.65	1332.60
MATERIAL											
RAW AND PURCH	66.66	79.06	112.65	135.67	158.12	154.59	151.95	149.85	148.10	146.61	1313.25
PURCHASED EQUIP	123.79	175.40	209.00	251.95	293.65	287.10	282.20	278.29	275.04	272.27	2438.90
TOTAL	190.45	254.47	321.65	387.61	451.78	441.70	434.15	428.13	423.14	418.87	3752.15
MISCELLANEOUS											
HOURS	1767.	1522.	1616.	1711.	1808.	1639.	1520.	1431.	1360.	1301.	15668.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	27.96	24.10	25.56	27.10	28.64	25.96	24.08	22.66	21.54	20.61	248.19
ENGINES	216.42	250.64	306.96	355.44	401.73	383.53	370.18	359.71	351.14	343.92	3347.66
AVIONICS	6.00	9.00	12.00	15.00	18.00	18.00	18.00	18.00	18.00	18.00	150.00
PROFIT	186.41	174.13	192.14	211.02	229.34	212.70	200.97	192.06	184.95	179.07	1962.79
INSUR.+TAXES	124.77	116.08	128.10	140.66	152.89	141.80	133.98	128.04	123.30	119.38	1308.53
WARRANTY	62.14	58.04	64.05	70.34	76.45	70.90	66.99	64.02	61.65	59.69	654.26
TOTAL FLYAWAY	1839.95	1778.74	1984.21	2199.08	2407.36	2244.92	2129.96	2042.24	1972.01	1919.74	20514.41

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COST SUMMARY

RDT AND E	TOTAL*
PROTOTYPE AIRCRAFT	916.66
DESIGN ENGINEERING	902.62
DEVELOPMENT TEST ARTICLES	421.17
FLIGHT TEST	116.33
ENGINE DEVELOPMENT CRUISE	956.61
ENGINE DEVELOPMENT LIFT	0.0
AVIONICS DEVELOPMENT	0.0
MAINTENANCE TRAINER LEVEL	0.0
OPERATOR TRAINER DEVELOP	0.0
DEVELOPMENT TOOLING	642.97
SPECIAL SUPPORT EQUIPMENT	16.32
DEVELOPMENT SPARES	139.16
TECHNICAL DATA	21.57
TOTAL RDTE	4334.61

INVESTMENT	TOTAL*	PER PROD A/C**
PRODUCTION AIRCRAFT	20514.41	68361.31
PRODUCTION ENGINEERING	0.0	0.0
MAINTENANCE TRAINERS	0.0	0.0
OPERATOR TRAINERS	0.0	0.0
PRODUCTION TOOLING	433.61	1445.36
SPECIAL SUPPORT EQUIPMENT	641.09	2803.63
PRODUCTION SPARES	3012.09	10040.30
TECHNICAL DATA	124.01	413.35
TOTAL INVESTMENT	24925.19	83063.94

DIRECT OPERATIONAL COST (DOC)	C/SM***	PERCENT
FLIGHT CREW	0.08881	4.60115
FUEL AND OIL	0.58421	30.26726
INSURANCE	0.18495	9.58196
DEPRECIATION	0.59507	30.83002
MAINTENANCE	0.47713	24.71965
TOTAL DOC	1.93016	100.000

INDIRECT OPERATIONAL COST (IOC)	C/SM***	PERCENT
SYSTEM	0.00383	0.42532
LOCAL	0.19273	21.40611
AIRCRAFT CONTROL	0.00513	0.56958
CABIN ATTENDANT	0.06839	7.59527
FOOD AND BEVERAGE	0.02363	2.62480
PASSENGER HANDLING	0.13656	15.16662
CARGO HANDLING	0.00849	0.94276
OTHER PASSENGER EXPENSE	0.33550	37.26233
OTHER CARGO EXPENSE	0.00278	0.30851
GENERAL + ADMINISTR.	0.12334	13.69868
TOTAL IOC	0.90037	100.000

MISC. DATA	
RANGE (ST. MILES)	4633.16
BLOCK SPEED (MPH)	1349.61
FARE ($)	248.73
FLEET SIZE	13.96
PRODUCTION BASIS	300.00
REV.PASSENG.(MIL.PER YR)	1.61
AVER. CARGO PER FLIGHT	2000.00
FLIGHT PER A/C PER YEAR	1005.41

RETURN ON INVESTMENT (ROI)	
TOTAL REVENUE PER YEAR *	469.74
TOTAL EXPENSE PER YEAR *	449.40
TOTAL INVESTMENT * INCL. FACILITIES	1301.65
ROI BEFORE TAXES	2.99
ROI AFTER TAXES	1.55

* - MILLIONS OF DOLLARS
** - 1000 OF DOLLARS PER PRODUCTION A/C
*** - CENTS PER SEAT MILE

RESEARCH DEVELOPMENT TEST AND EVALUATION (RDTE)

	DEVELOPMENT AND DESIGN	CONTRACTOR TEST AND EVALU	DEVELOPMENT AIRCRAFT	TOTAL RDT AND E
AIRFRAME	1517.91	467.40	636.94	2622.24
ENGINEERING				
HOURS	45186.	10146.	3145.	58476.
LABOR RATE	8.17	8.17	8.17	8.17
OVERHEAD RATE	9.20	9.20	9.20	9.20
TOTAL	784.89	176.24	54.63	1015.76
TOOLING				
HOURS	36324.	2621.	5242.	44187.
LABOR RATE	6.09	6.09	6.09	6.09
OVERHEAD RATE	12.36	12.36	12.36	12.36
TOTAL	733.02	48.36	96.71	878.08
MANUFACTURING				
HOURS		10484.	20967.	31451.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		166.06	332.12	498.18
QUALITY CONTROL				
HOURS		2097.	4193.	6290.
LABOR RATE		6.29	6.29	6.29
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		35.67	71.33	107.00
MATERIAL				
RAW AND PRCHSD		12.05	24.10	36.15
PURCHASED EQUIP		22.38	44.76	67.14
TOTAL		34.43	68.86	103.29
MISCELLANEOUS				
HOURS		419.	839.	1258.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		6.64	13.28	19.93
ENGINES	956.61		86.04	1042.65
AVIONICS	0.0		2.00	2.00
PROFIT(AIRFRAME)	227.69	70.11	95.54	393.34
INSUR.+TAXES			63.69	63.69
WARRANTY			31.85	31.85
SUBTOTAL	2702.20	537.51	916.06	4155.76
OTHER ITEMS				179.05
TOTAL (RDTE)				4334.81

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 SUMMARY ID NO.  165                    A S S E T   P A R A M E T R I C   A N A L Y S I S                  FEBRUARY 27 1974

AIRCRAFT MODEL --CL 1607-5                          ENGINE I.D. -- 97000                WING QUARTER CHORD SWEEP = 68.63 DEG
I.O.C. DATE    --1990                               SLS SCALE 1.0 =  78000              WING TAPER RATIO = 0.0
DESIGN SPEED   --SUPERSONIC                         NUMBER OF ENGINES = 4.

  1 W/S               69.3   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  2 T/W              0.477   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  3 AR                1.62   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  4 T/C               2.65   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  5 RADIUS N. MI      4200     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0

  6 GROSS WEIGHT    759498     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
  7 FUEL WEIGHT     394017     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
  8 OP. WT. EMPTY   316481     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
  9 ZERO FUEL WT.   365481     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
 10 THRUST/ENGINE    90559     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
 11 ENGINE SCALE     1.162   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 12 WING AREA       10959.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.
 13 WING SPAN        133.3   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 14 H. TAIL AREA    1106.7   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 15 V. TAIL AREA     471.9   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 16 BODY LENGTH      297.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
COST DATA
 17 RDTE - BIL.      4.335   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 18 FLYAWAY - MIL.   44.74   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 19 INVESTMNT-BIL.   1.362   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 20 DOC - C/SM       1.930   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 21 IOC - C/SM       0.960   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 22 ROI A.T. - O/O    1.55   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
CONSTRAINT OUTPUT
 23 TAKEOFF DST(1)    8870     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
 24 CLIMB GRAD(1)   0.2265   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 25 TAKEOFF DST(2)    6736     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
 26 CLIMB GRAD(2)   0.1068   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 27 CTOL LNDG D(1)    8858     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
 28 AP SPEED-KT(1)   149.1   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 29 CTOL LNDG D(2)    9100     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
 30 AP SPEED-KT(2)   151.7   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
 31 CTOL LNDG D(3)    9336     0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
 32 AP SPEED-KT(3)   154.2   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
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                                  INTERNATIONAL MISSION  STD DAY + 14.4 F

                                                    T/C     AR    W/S    T/W
--------------------------------------------------------------------------------------------
                                                    2.65   1.62   69.3   0.477


                                        W E I G H T     S T A T E M E N T
--------------------------------------------------------------------------------------------

                                               WEIGHT(POUNDS)      WEIGHT FRACTION        (PERCENT)

   TAKE-OFF WEIGHT                            (   750000.)
   FUEL AVAILABLE                                 387875.          FUEL                      51.72
   ZERO FUEL WEIGHT                           (   362125.)
   PAYLOAD                                         49000.          PAYLOAD                    6.53
   OPERATING WEIGHT                           (   313125.)
      OPERATING ITEMS                               5473.          OPERATING ITEMS            1.44
      STANDARD ITEMS                                5329.
   EMPTY WEIGHT                               (   302323.)
      WING                                         89434.
      TAIL                                         11339.
      BODY                                         42688.          STRUCTURE                 25.06
      LANDING GEAR                                 30401.
      SURFACE CONTROLS                              8500.
      NACELLE AND ENGINE SECTION                    5616.
      PROPULSION                              (    79165.)         PROPULSION                10.56
         WEIGHT OF LIFT ENGINES                        0.
         VECTOR CONTROL SYSTEM                         0.
         ENGINES                                   51124.
         THRUST REVERSAL                               0.
         AIR INDUCTION SYSTEM                      20755.
         FUEL SYSTEM                                5676.
         ENGINE CONTROLS + STARTER                  1631.
      INSTRUMENTS                                   1230.
      HYDRAULICS                                    5700.
      ELECTRICAL                                    4550.
      AVIONICS                                      1900.          EQUIPMENT                  4.69
      FURNISHINGS AND EQUIPMENT                    11500.
      ENVIRONMENTAL CONTROL SYSTEM                  8300.
      AUXILIARY GEAR                                1960.

   A.M.P.R.                                   (   233942.)         TOTAL                    100.00)
--------------------------------------------------------------------------------------------
   EXCESS FUEL CAPACITY - BODY                    838211.
   EXCESS FUEL CAPACITY - WING                         0.
   EXCESS BODY LENGTH - FT                           0.0

   STRUCTURE   ALUMINUM
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                              W E I G H T S   M A T R I X

   / MATERIAL
ELEMENT/                     AL      TIT.     STEEL     COMP.     OTHER     TOTAL

               WING         4114.   76555.    1789.     5545.     1431.    89434.
               TAIL          510.   10534.     113.        0.      181.    11339.
               FUSEL        1964.   33895.     768.     1067.     4995.    42688.
               L. G.          30.    7600.   11674.        0.    11096.    30401.
               NACELLE       107.     834.    1667.        0.        0.     2608.
               AIR INDUCT    955.   18389.     208.        0.     1204.    20755.
               S. CTLS      2040.     363.    1785.      126.     4165.     8500.

               TOTALS       9720.  146189.   18204.     6740.    23072.   205925.

                        C O N F I G U R A T I O N   G E O M E T R Y

BASIC WING--      AREA(SQ.FT)   SPAN(FT)  TAPER RATIO  C/4 SWEEP  L.E. SWEEP  CR(FT)      MAC(FT)
                    10622.0      132.50      0.0        68.626      72.500    163.35      108.91

INBOARD WING--    AREA(SQ.FT)   EXP. AREA L.E. SWEEP   REF L(FT)  SFLE(SQ.FT) AVG T/C
                    10622.0      9330.0      72.50      101.12       0.0        2.65

OUTBOARD WING--   AREA(SQ.FT)   Y BRK(FT) L.E. SWEEP   REF L(FT)  SFLE(SQ.FT) AVG T/C
                        0.0        0.0       72.50      101.12       0.0        2.65

TOTAL WING--      AREA(SQ.FT)    EFF AR    AVG T/C      CR(FT)      CT(FT)   (B/2)/LW      P
                    10622.0       1.62       2.65       163.35        0.0      0.315       0.389

WING TANK--        CBAR1(FT)   CBAR2(FT)    FTL(FT)  FVWING(CU FT)  FVBOX(CU FT)
                     149.38       0.0        60.56     4462.63      2763.53

FUSELAGE--        LENGTH(FT)  S WET(SQ FT)   BWR(FT)   EQUIV D(FT)  SPI(SQ FT)
                     297.00     7876.0         9.47       13.22      137.30

                     HW(FT)      BH(FT)   SBW(SQ FT)   FVB(CU FT)
                     11.33       11.50     9177.00     20000.00

TAIL--            SHT(SQ.FT)  SHTX(SQ.FT) HT REF L(FT) SVT(SQ.FT)  SVTX(SQ.FT)  VT REF L(FT)
                    1065.00     914.00       23.50      466.00      406.00       25.66

PROPULSION--       ENG L(FT)   ENG D(FT)   POD L(FT)   POD D(FT)   POD S WET    NO. PODS  INLET L(FT)
                     22.29        7.09       39.19        8.03      3955.73         4.        0.0
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M I S S I O N   S U M M A R Y

INTERNATIONAL MISSION STD LAY + 14.4 F

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG L/D RATIO	AVG SFC (FF/T)	MAX OVER PRES
TAKEOFF															
POWER 1	0.	0.0	750000.	3569.	3569.	0.	0.	10.0	10.0	0.	-97101.	0.	0.0	0.571	0.0
POWER 2	0.	0.300	746431.	6036.	9605.	0.	0.	0.5	10.5	0.	97201.	0.	5.18	1.388	0.0
CLIMB	0.	0.300	740395.	7172.	16777.	4.	4.	1.1	11.5	0.	97201.	0.	7.45	1.446	0.0
CRUISE	5000.	0.413	733223.	4299.	21076.	0.	4.	4.0	15.5	0.	-97101.	0.	6.24	0.727	0.0
ACCEL	5000.	0.413	728924.	1771.	22847.	1.	6.	0.3	15.8	0.	97201.	0.	9.96	1.472	0.0
CLIMB	5000.	0.537	727153.	22508.	45355.	36.	42.	4.7	20.5	0.	97201.	0.	11.25	1.508	0.0
CLIMB	34000.	0.964	704644.	49335.	94690.	255.	297.	14.3	34.8	0.	97201.	0.	8.32	1.533	0.0
CLIMB	63000.	2.603	655310.	0.	94690.	0.	297.	0.0	34.8	0.	97201.	0.	8.73	1.563	0.0
CRUISE	63000.	2.620	655310.	224611.	319301.	3693.	3990.	144.7	179.5	0.	-97201.	0.	8.61	1.498	0.0
DECEL	73000.	2.620	430699.	96.	319396.	23.	4013.	0.9	180.4	0.	97501.	0.	6.42	-0.769	0.0
DESCENT	73000.	2.309	430603.	1281.	320678.	154.	4166.	13.1	193.5	0.	97501.	0.	9.15	-0.386	0.0
RESET	5000.	0.0	429322.	0.	320678.	0.	4166.	0.0	193.5	0.	0.	0.	0.0	0.0	0.0
CRUISE	5000.	0.413	429322.	2643.	323321.	0.	4166.	5.0	198.5	0.	-97101.	0.	11.17	0.829	0.0
RESET	0.	0.0	426679.	0.	323321.	0.	4166.	0.0	198.5	0.	0.	0.	0.0	0.0	0.0
RESET	0.	0.0	426679.	0.	323321.	-4166.	0.	*****	0.0	0.	0.	0.	0.0	0.0	0.0
RESERVE	0.	0.0	426679.	22632.	345953.	0.	0.	0.0	0.0	0.	0.	0.	0.0	0.0	0.0
CLIMB	0.	0.200	404047.	2751.	348704.	2.	2.	0.4	0.4	0.	97201.	0.	9.71	1.444	0.0
CLIMB	1500.	0.501	401295.	13320.	362024.	22.	23.	3.0	3.4	0.	97201.	0.	9.85	1.504	0.0
CRUISE	43000.	0.900	387976.	10402.	372426.	167.	190.	19.0	22.4	0.	-97101.	0.	10.93	0.937	0.0
DESCENT	43000.	0.900	377573.	774.	373200.	56.	246.	7.9	30.3	0.	97501.	0.	9.89	-0.491	0.0
CRUISE	43000.	0.900	376800.	756.	373957.	12.	260.	1.4	31.7	0.	-97101.	0.	10.92	0.938	0.0
LOITER	15000.	0.509	376042.	13910.	387867.	0.	260.	30.0	61.7	0.	-97101.	0.	11.29	0.851	0.0

TOGWT= 750000.0   FUEL A=367674.7   FUEL R=387867.6

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PRODUCTION

	PRODUCTION YEARS										
	1	2	3	4	5	6	7	8	9	10	TOTAL
AIRFRAME	1229.36	1148.43	1267.29	1391.85	1512.85	1402.92	1325.60	1266.62	1219.90	1181.15	12945.97
ENGINEERING											
HOURS	4366.	3763.	3982.	4231.	4472.	4053.	3760.	3538.	3362.	3217.	38746.
LABOR RATE	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	
OVERHEAD RATE	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	
TOTAL	75.84	65.36	69.16	73.50	77.68	70.40	65.31	61.46	58.40	55.88	673.01
TOOLING											
HOURS	5242.	4515.	4778.	5077.	5366.	4864.	4512.	4246.	4034.	3860.	46495.
LABOR RATE	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	
OVERHEAD RATE	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	
TOTAL	96.72	83.31	88.15	93.66	99.01	89.73	83.24	78.33	74.43	71.23	857.83
MANUFACTURING											
HOURS	43684.	37628.	39817.	42312.	44718.	40530.	37599.	35382.	33620.	32171.	387458.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	691.95	596.02	630.70	670.22	708.33	642.00	595.56	560.44	532.54	509.58	6137.33
QUALITY CONTROL											
HOURS	8737.	7526.	7963.	8462.	8944.	8106.	7520.	7076.	6724.	6434.	77492.
LABOR RATE	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	148.61	128.01	135.46	143.94	152.13	137.88	127.91	120.37	114.37	109.44	1318.13
MATERIAL											
RAW AND PURCH	65.98	88.16	111.51	134.29	156.52	153.03	150.41	148.33	146.60	145.12	1299.96
PURCHASED EQUIP	122.54	163.73	207.08	249.40	290.68	284.20	279.34	275.47	272.25	269.51	2414.21
TOTAL	188.53	251.89	318.59	383.69	447.20	437.23	429.76	423.80	418.85	414.63	3714.17
MISCELLANEOUS											
HOURS	1747.	1555.	1593.	1692.	1769.	1621.	1504.	1415.	1345.	1287.	15498.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	27.68	24.64	25.23	26.81	26.33	25.68	23.82	22.42	21.30	20.38	245.49
ENGINES	216.78	254.71	304.65	352.76	398.71	360.64	367.40	357.01	348.50	341.33	3322.48
AVIONICS	6.00	9.00	12.00	15.00	18.00	18.00	18.00	18.00	18.00	18.00	150.00
PROFIT	184.40	172.28	190.09	208.77	226.90	210.44	198.84	190.02	182.98	177.17	1941.90
INSUR.+TAXES	122.94	114.84	126.73	139.18	151.27	140.29	132.58	126.68	121.99	118.12	1294.60
WARRANTY	61.47	57.42	63.36	69.59	75.63	70.15	66.26	63.34	60.99	59.06	647.30
TOTAL FLYAWAY	1820.95	1750.67	1964.12	2177.14	2383.19	2222.44	2108.68	2021.87	1952.37	1900.64	20308.05

COST SUMMARY

RDT AND E		INVESTMENT			DIRECT OPERATIONAL COST (DOC)		
	TOTAL*		TOTAL*	PER PROD A/C**		C/SM***	PERCENT
PROTOTYPE AIRCRAFT	906.42	PRODUCTION AIRCRAFT	20308.03	67693.44	FLIGHT CREW	0.08896	4.64209
DESIGN ENGINEERING	695.10	PRODUCTION ENGINEERING	0.0	0.0	FUEL AND OIL	0.57925	30.21986
DEVELOPMENT TEST ARTICLES	416.65				INSURANCE	0.16351	9.57358
FLIGHT TEST	114.89				DEPRECIATION	0.59043	30.80304
ENGINE DEVELOPMENT CRUISE	950.01				MAINTENANCE	0.47462	24.76144
ENGINE DEVELOPMENT LIFT	0.0				TOTAL DOC	1.91679	100.000
AVIONICS DEVELOPMENT	0.0						
MAINTENANCE TRAINER DEVEL	0.0	MAINTENANCE TRAINERS	0.0	0.0	INDIRECT OPERATIONAL COST (IOC)		
OPERATOR TRAINER DEVELOP	0.0	OPERATOR TRAINERS	0.0	0.0		C/SM***	PERCENT
DEVELOPMENT TOOLING	636.37	PRODUCTION TOOLING	432.32	1441.06	SYSTEM	0.00381	0.42359
SPECIAL SUPPORT EQUIPMENT	18.13	SPECIAL SUPPORT EQUIPMENT	832.63	2775.43	LOCAL	0.19186	21.30754
DEVELOPMENT SPARES	157.81	PRODUCTION SPARES	2983.64	9945.46	AIRCRAFT CONTROL	0.00517	0.57414
TECHNICAL DATA	21.36	TECHNICAL DATA	122.78	409.26	CABIN ATTENDANT	0.06852	7.60923
					FOOD AND BEVERAGE	0.02366	2.62962
TOTAL RDTE	4296.60	TOTAL INVESTMENT	24679.41	82264.69	PASSENGER HANDLING	0.13766	15.28798
MISC. DATA		RETURN ON INVESTMENT (ROI)			CARGO HANDLING	0.00856	0.95030
RANGE (ST. MILES)	4794.47	TOTAL REVENUE PER YEAR *		466.11	OTHER PASSENGER EXPENSE	0.33550	37.25967
BLOCK SPEED (MPH)	1347.24	TOTAL EXPENSE PER YEAR *		443.71	OTHER CARGO EXPENSE	0.00278	0.30849
FARE ($)	246.81	TOTAL INVESTMENT * INCL. FACILITIES		1340.40	GENERAL + ADMINISTR.	0.12290	13.64942
FLEET SIZE	13.88	ROI BEFORE TAXES		3.34	TOTAL IOC	0.90044	100.000
PRODUCTION BASIS	300.00	ROI AFTER TAXES		1.74			
REV.PASSENG.(MIL.PER YR)	1.61						
AVER. CARGO PER FLIGHT	2000.00						
FLIGHT PER A/C PER YEAR	1011.59						

* - MILLIONS OF DOLLARS
** - 1000 OF DOLLARS PER PRODUCTION A/C
*** - CENTS PER SEAT MILE

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RESEARCH DEVELOPMENT TEST AND EVALUATION (RDTE)

	DEVELOPMENT AND DESIGN	CONTRACTOR TEST AND EVALU	DEVELOPMENT AIRCRAFT	TOTAL RDT AND E
AIRFRAME	1505.63	462.19	630.07	2597.90
ENGINEERING				
HOURS	44810.	10027.	3111.	57948.
LABOR RATE	8.17	8.17	8.17	8.17
OVERHEAD RATE	9.20	9.20	9.20	9.20
TOTAL	778.35	174.17	54.04	1006.56
TOOLING				
HOURS	36040.	2592.	5185.	43617.
LABOR RATE	6.09	6.09	6.09	6.09
OVERHEAD RATE	12.36	12.36	12.36	12.36
TOTAL	727.28	47.83	95.66	670.77
MANUFACTURING				
HOURS		10370.	20740.	31109.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		164.26	326.51	492.77
QUALITY CONTROL				
HOURS		2074.	4148.	6222.
LABOR RATE		6.29	6.29	6.29
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		35.28	70.56	105.83
MATERIAL				
RAW AND PRCHSL		11.93	23.86	35.79
PURCHASED EQUIP		22.15	44.31	66.46
TOTAL		34.08	66.16	102.25
MISCELLANEOUS				
HOURS		415.	830.	1244.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		6.57	13.14	19.71
ENGINES	950.01		85.39	1035.41
AVIONICS	0.0		2.00	2.00
PROFIT(AIRFRAME)	225.84	69.33	94.51	389.68
INSUR.+TAXES			63.01	63.01
WARRANTY			31.50	31.50
SUBTOTAL	2681.49	531.52	906.48	4119.49
OTHER ITEMS				177.31
TOTAL (RDTE)				4296.80

```
  SUMMARY ID NO. 166                  A S S E T   P A R A M E T R I C   A N A L Y S I S                     FEBRUARY 27 1974

AIRCRAFT MODEL --CL 1607-5                           ENGINE I.D. -- 97000                  WING QUARTER CHORD SWEEP = 68.63 DEG
I.O.C. DATE    --1980                                SLS SCALE 1.0 =  76000                WING TAPER RATIO = 0.0
DESIGN SPEED   --SUPERSONIC                          NUMBER OF ENGINES = 4.

   1 W/S                69.2    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
   2 T/W               0.477    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
   3 AR                 1.62    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
   4 T/C                4.65    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
   5 RADIUS N. MI       4100      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0

   6 GROSS WEIGHT     750000      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
   7 FUEL WEIGHT      387874      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
   8 OP. WT. EMPTY    313125      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
   9 ZERO FUEL WT.    362125      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
  10 THRUST/ENGINE     69465      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
  11 ENGINE SCALE      1.147    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  12 WING AREA        10622.     0.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.    0.
  13 WING SPAN         132.3    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  14 H. TAIL AREA     1065.0    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  15 V. TAIL AREA      466.0    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  16 BODY LENGTH       297.0    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
COST DATA
  17 RDTE - BIL.       4.297    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  18 FLYAWAY - MIL.    95.61    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  19 INVESTMNT-BIL.    1.540    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  20 DOC - C/SM        1.917    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  21 IOC - C/SM        0.961    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  22 ROI A.T. - O/O     1.74    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
CONSTRAINT OUTPUT
  23 TAKEOFF DST(1)     8870      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
  24 CLIMB GRAD(1)    0.2265    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  25 TAKEOFF DST(2)     8736      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
  26 CLIMB GRAD(2)    0.1066    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  27 CTOL LNDG D(1)     6879      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
  28 AP SPEED-KT(1)    149.3    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  29 CTOL LNDG D(2)     9124      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
  30 AP SPEED-KT(2)    151.9    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
  31 CTOL LNDG D(3)     9365      0     0     0     0     0     0     0     0     0     0     0     0     0     0     0
  32 AP SPEED-KT(3)    154.5    0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0   0.0
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SECTION 18

DESIGN

BY

C.W. LINDBLOM

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SECTION 18

DESIGN

INTRODUCTION

An important facet of the structural design concepts study is the establishment of a firm design technology base that will permit the best possible judgement to be exercised in the selection of structural concepts and materials for the next generation supersonic transport. This design technology base was determined with the full consideration of:

- the practically obtainable benefits of applicable advanced technology
- the internal structural response to the complex interactions among inertial, aerodynamics and thermal loadings
- the effects of various structural arrangements, concepts and materials on these interactions

An underlying objective was the achievement of the minimum possible structural mass fraction since the economic effectiveness of the supersonic cruise aircraft is critically sensitive to inert mass as well as aerodynamic and propulsive efficiencies.

In approaching this task the following were identified as encompassing the potential for structural mass reduction:

- Improved titanium alloys (beta alloys)
- Improved fatigue quality through minimizing fasteners by the use of welding, bonding, brazing, weld bonding, weld brazing and rivet bonding
- Large scale fabrication to minimize the number of joints
- Minimizing or eliminating tank sealing by the use of large scale application of welding, bonding and brazing
- Selective reinforcing of metal structure with organic and metal matrix composites

- Determining the structural arrangements most efficient in coping with the interactive loading of a large, flexible arrow-wing
- Determining the most efficient structural concepts within these structural arrangements

The initial design effort was devoted to the establishment of the baseline configuration concepts and the subsequent development of internal details to provide inputs to mass distribution and other studies included in the structural study effort. The various drawings developed are presented in Section 2, Baseline Configuration Concept. The development of these drawings made full use of the information of Reference 1.

In developing the structural arrangement drawings for the various wing and fuselage design approaches, an extensive background of design data was available from the design and manufacturing studies of advanced structural designs having potential application in a Mach 2.7 arrow-wing configuration transport. These studies included numerous consultations with vendors (Table 18-1) and other contractors, extensive literature surveys and close collaboration with producibility personnel. This background data was fully utilized in the structural arrangement drawings discussed in the subsequent sections as well as in the determination of the design parameters (Section 8).

TABLE 18-1. VENDOR CONTACTS MADE FOR ARROW WING STRUCTURES STUDY

Vendor	Subject Discussed
Aeronca Mfg.	Aluminum Brazed Honeycomb Sandwich
Rohr Aircraft	Liquid Interface Diffusion (LID) Diffusion Braze Process
Northrop Aircraft	Nor-Ti-Bond Diffusion Braze Process
TRW Systems	Weld Bond
Advanced Structures and Technology Co.	STRESSKIN
Sciaky Brothers, Inc.	Welding
Avco	Metal Matrix Composites
Holosonics	NDT Inspection Methods

DESIGN APPROACH

The various approaches in the arrangement of surface panel concepts and structure for primary structure design are presented in Section 1, Structural Design Concepts. The evaluation rationale of the structural approaches was also presented (Table 18-2). Each step of the procedure was outlined from:

(1) Establishing the material system

(2) Defining the fabrication and/or assembly method for the candidate concepts

(3) Conducting the design analysis according to the guidelines defined

(4) Establishing mass and cost data, and

(5) Evaluating the results and selecting the most promising arrangement for further detailed study.

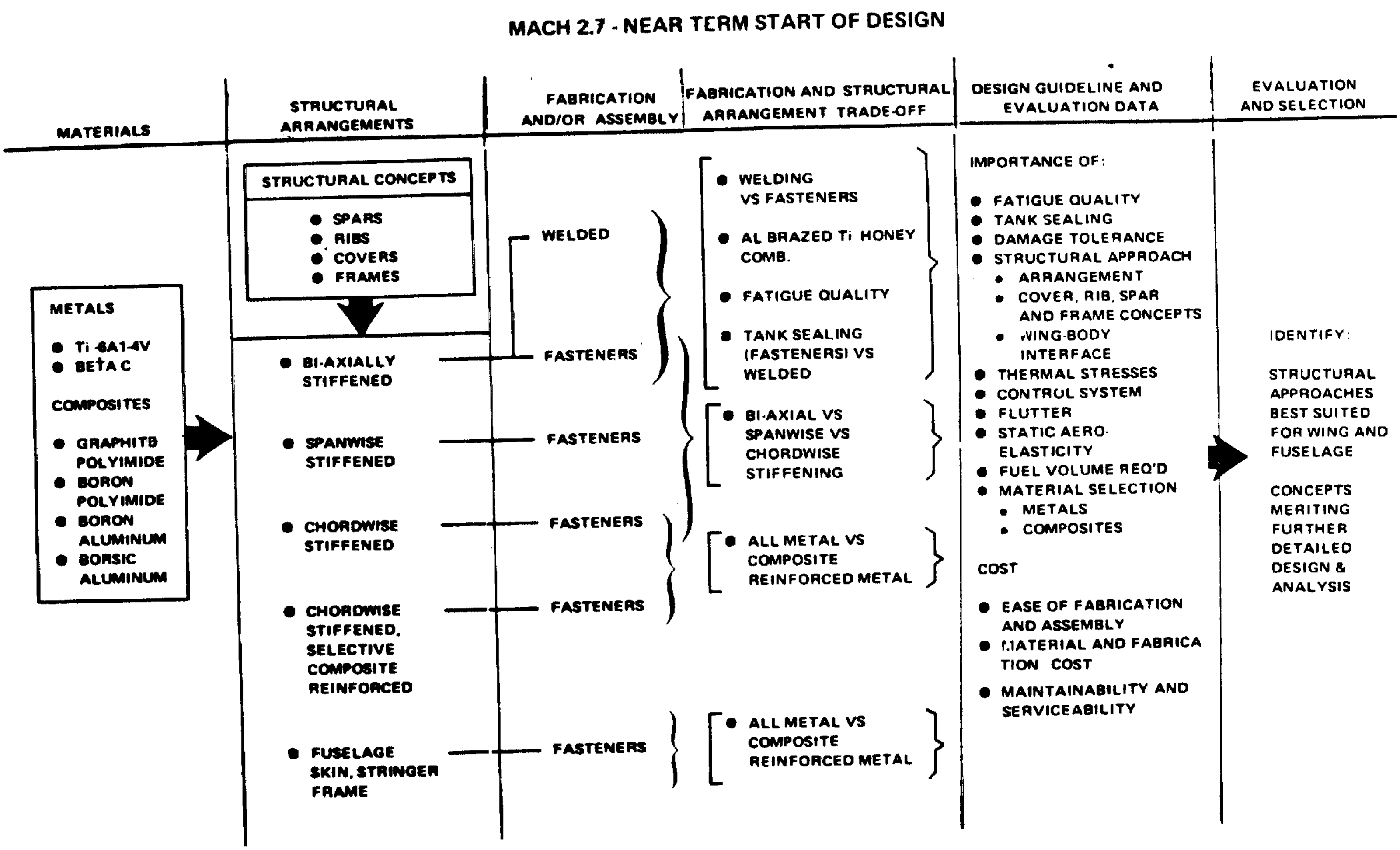
Joining Methods

A vital part of the study matrix (Table 18-2) is the evaluation of the joining methods. These are important parameters because of their influence on such factors as:

- Fatigue quality and damage tolerance
- Manufacturing cost, facilities and techniques
- Serviceability and maintainability
- Fuel tank sealing
- Thermal stresses, residual stresses, and mass

Based on the comprehensive investigation into the various joining methods and their applicability to the candidate structural design concepts, two methods were selected as identified in the Fabrication and/or Assembly column of Table 18-2.

TABLE 18-2. EVALUATION OF STRUCTURAL APPROACHES — TASK I



MACH 2.7 - NEAR TERM START OF DESIGN

MATERIALS	STRUCTURAL ARRANGEMENTS	FABRICATION AND/OR ASSEMBLY	FABRICATION AND STRUCTURAL ARRANGEMENT TRADE-OFF	DESIGN GUIDELINE AND EVALUATION DATA	EVALUATION AND SELECTION
METALS ● Ti-6A1-4V ● BETA C COMPOSITES ● GRAPHITE POLYIMIDE ● BORON POLYIMIDE ● BORON ALUMINUM ● BORSIC ALUMINUM	STRUCTURAL CONCEPTS ● SPARS ● RIBS ● COVERS ● FRAMES			IMPORTANCE OF: ● FATIGUE QUALITY ● TANK SEALING ● DAMAGE TOLERANCE ● STRUCTURAL APPROACH • ARRANGEMENT • COVER, RIB, SPAR AND FRAME CONCEPTS • WING-BODY INTERFACE ● THERMAL STRESSES ● CONTROL SYSTEM ● FLUTTER ● STATIC AERO-ELASTICITY ● FUEL VOLUME REQ'D ● MATERIAL SELECTION • METALS • COMPOSITES COST ● EASE OF FABRICATION AND ASSEMBLY ● MATERIAL AND FABRICATION COST ● MAINTAINABILITY AND SERVICEABILITY	IDENTIFY: STRUCTURAL APPROACHES BEST SUITED FOR WING AND FUSELAGE CONCEPTS MERITING FURTHER DETAILED DESIGN & ANALYSIS
	● BI-AXIALLY STIFFENED	WELDED FASTENERS	● WELDING VS FASTENERS ● AL BRAZED Ti HONEY COMB. ● FATIGUE QUALITY ● TANK SEALING (FASTENERS) VS WELDED		
	● SPANWISE STIFFENED	FASTENERS	● BI-AXIAL VS SPANWISE VS CHORDWISE STIFFENING		
	● CHORDWISE STIFFENED	FASTENERS	● ALL METAL VS COMPOSITE REINFORCED METAL		
	● CHORDWISE STIFFENED, SELECTIVE COMPOSITE REINFORCED	FASTENERS			
	● FUSELAGE SKIN, STRINGER FRAME	FASTENERS	● ALL METAL VS COMPOSITE REINFORCED METAL		

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- The "fastener" joining method is taken as a baseline case and is investigated for all wing and fuselage structural arrangements.
- The two joining methods (welding and fasteners) are considered for the monocoque wing arrangements.

The logic behind this reduced matrix is that the study of the two methods noted above will yield sufficient data concerning the merits of each fabrication and/or assembly method to permit valid projections to be made concerning the effects of joining methods on the spanwise and chordwise stiffened wing arrangements without the need for a design development effort for each individual joining method. Titanium alloy Ti 6Al-4V is taken as the baseline metal for the design study even though Beta C and other titanium alloys were investigated.

As a matter of clarification, the joining methods described herein are concerned with methods used to assemble major components (wing surface assemblies, spar assemblies, etc.) to their next assembly. It does not imply that all attachments within these major components use the same joining method. For example:

- A wing structure design using honeycomb sandwich surface panels attached to substructure by fasteners is included in the "fastener" assembly concept even though the surface panels themselves might be fabricated as aluminum brazed, bonded or welded honeycomb sandwiches.
- A fuselage structure design with skin panel assemblies joined together by fasteners also is classified under the "fastener" concept even though stringers, frame segments and other detail parts of skin panel assemblies are attached by weld bonding, brazing or other means.

Manufacturing/Subcontracting/Facilities Guidelines

The focus on the method of joining major components reflects the complexity required in manufacturing equipment, facilities and techniques. For example, the "fastener" concept permits conventional manufacturing approaches to be utilized. In contrast, the "welding" designation indicates that complex and expensive facilities and equipment may be required for manufacturing and maintenance. These factors relate to a number of items, including the feasibility of a particular "design go-ahead" date.

The interpretations above permit subassemblies to utilize advanced assembly processes, such as weld bonding, weld brazing and others, consistent with the assumed "design go-ahead" and the fabrication and transportation size limits established.

Guidelines in reference to manufacturing, subcontracting and facilities were established considering the above to aid in the design process. The guidelines, discussed in Section 7, Materials and Producibility and Section 8, Basic Design Parameters, are summarized below:

- The entire fuselage, except for the flight station, is designed for subcontracting (and transportation limitations).
- All wing segments, and the complete wing, to be assembled in a new facility assumed to be constructed in Palmdale. Wing components, such as spars and ribs, to be suitable for subcontracting.
- The new Palmdale facility is postulated to have autoclave and fusion and spot welding equipment suitable for fabricating large components.
- Fuselage segment joints need not coincide with wing segment joints.

Design Ground Rules and Constraints

The ground rules enumerated below were established at the beginning of the study and are presented to define the basis for the design effort. Some of these rules might be modified by new developments or as a result of further study. Specific guidelines for manufacturing are presented in Section 7, Materials and Producibility.

- Minimum mass is the primary design objective.
- Design work is initially based on all-metal structure reflecting the technology level for a design go-ahead at the end of 1975. This technology level is expanded to include selective reinforcing of the metal structure with composites considering a design go-ahead in 1981.
- Bonding, weld bonding, rivet bonding, weld brazing and the aluminum braze (Aeronca) processes are postulated to be developed adequately and available for use.
- TIG, plasma arc and EB welding acceptable but laser welding not yet qualified.
- Component size limits, as dictated by fabricating processes, to be:
  - o Hot vacuum forming - 15 feet wide by 35 feet long
  - o Aluminum brazed sandwich - 68 inch wide by 40 feet long
  - o Weld bonded and rivet bonded panels - 15 feet wide by 50 feet long
- Component size limits, as dictated by transportation method, defined in Section 7, Materials and Producibility.
- Structure to have an economic service life of 15 years, a service life of 50,000 flight hours, a design fatigue life of twice the service life and damage-tolerant structural concepts where possible.
- Structure design to be based on Mach 2.7 cruise speed.
- Fuel tank access to be provided by a minimum of 2 doors for each fuel tank or isolated portion thereof. Access opening size to be approximately 13 x 18 inches for vertical entry and 20 x 31 inches for horizontal entry.
- Minimum skin gauges (inch) to be:
  - o 0.020 for exterior skins on the wing lower surface
  - o 0.015 for exterior skin on the wing upper surface
  - o 0.010 for the interior face of a sandwich

- All fasteners in exterior surface to be flush.
- External splice strap acceptable at all longitudinal panel splices in the fuselage and at BL 470 joint in the wing.
- At countersunk fasteners, the face sheet thickness shall be at least 0.010 inches greater than the countersink depth to avoid "feather edges" and degraded quality. This rule might be waived in joints where the reduced allowables and degraded quality resulting from feather edges are acceptable.
- Frame locations different from spar locations are considered acceptable but will be used only where an overall advantage is indicated after evaluation of any extra material required to redistribute and/or account for loadings encountered at BL 62.41 rib.
- For welded/corrugated web spars and ribs, it is assumed that the cap strips should be normal to the plane of the web. (Note: subsequent meetings with Sciaky welding specialists indicate the feasibility of welding caps to webs at angles varying as much as 15-degrees from normal to the web).
- Manually upset Ti rivets (B-120 and Beta C) limited to 0.156 inch diameter maximum.
- All Ti Hi-Tigue fasteners to have studs of Ti-6Al-4V in the STA condition.
- Shear (shallow) head flush fasteners acceptable in the outer skin.
- Single row of fasteners unacceptable at skin panel joint if the joint must be fuel tight.
- For welded joints:
    - Manual welding limited to an absolute minimum
    - Skin splice welds planished
    - All welds to be the butt type
    - All welds shaved on all surfaces unless impossible
    - Welds are designed for stress relieving where feasible. However, welded spars and ribs will not be stress relieved. Note that welded joints in members which are later hot formed are stress-relieved automatically.

- In the primary structure, access doors are to be the "clamped" type where feasible to avoid fastener holes in primary structure.
- From the manufacturing standpoint, blind fasteners are preferred over access doors.

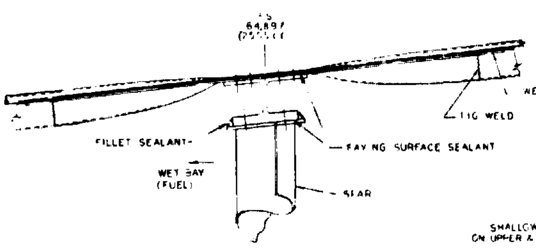
Further study and development may indicate the need for amplification and/or modification of the above rules.

STRUCTURAL DESIGN — TASK I

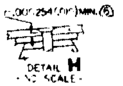
Chordwise Stiffened Wing Design — Fasteners

The structural arrangement drawing in the chordwise stiffened wing design is presented in Figure 18-1. The arrangement of the substructure, fuel tanks and surface panel joints for the representative panels are indicated on the airplane planform. The highly efficient convex beaded surface panel design is shown with discrete submerged spar caps used to transmit the wing spanwise bending loads. Representative spar cap and truss-web geometry are indicated for selected point design regions identified on the wing planform. General features for this design are enumerated below:

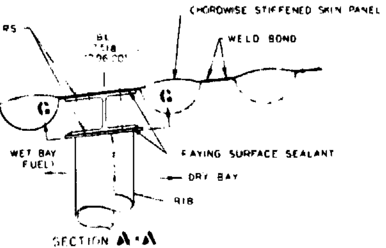
- Double-skin wing surfaces in which both the inner and outer skins contain a series of chordwise-oriented stiffening beads.
- Numerous spanwise spars which are of truss design except where a spar serves as a fuel tank wall or a fuel surge pressure spar. Where spar depth or loading so dictates, the truss spar design may be replaced by a stiffened web or other arrangement.
- At fuel tank wall and fuel baffle locations, the spars have welded corrugated webs and I-section caps.
- Relatively few chordwise ribs are used. These also are of the truss type design except where the rib serves as a tank wall or baffle. In such applications, welded corrugated webs and I-section caps are used.
- Spars are spaced relatively close in the main wing box behind the main landing gear (MLG) well.


FS
64.897
(2555.0)
FILLET SEALANT
WET BAY
(FUEL)
FAYING SURFACE SEALANT
SPAR
TIG WELD
WELD BOND DOUBLER

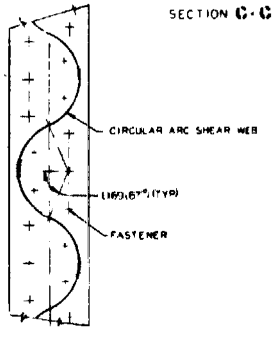
SECTION C-C


.0010(.254)(.010)MIN.

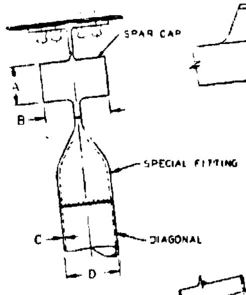
DETAIL H
NO SCALE


FASTENERS
BL
.118
(06.00)
CHORDWISE STIFFENED SKIN PANEL
WELD BOND
C
C
WET BAY
FUEL
FAYING SURFACE SEALANT
DRY BAY
RIB

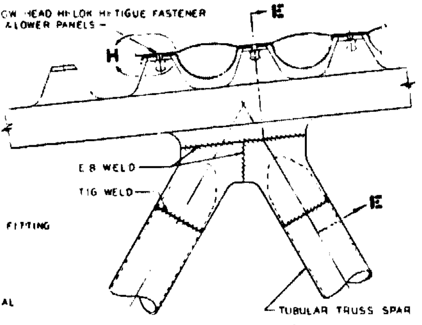
SECTION A-A
(ROTATED 90° CCW)


CIRCULAR ARC SHEAR WEB
1.169(.67°)(TYP)
FASTENER

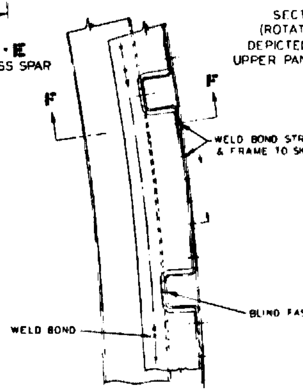
SECTION G-G


SPAR CAP
A
B
SPECIAL FITTING
C
DIAGONAL
D

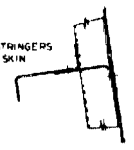
SECTION E-E
TUBULAR TRUSS SPAR


SHALLOW HEAD HI-LOK HI-TIGUE FASTENER
ON UPPER & LOWER PANELS
E
H
E.B WELD
TIG WELD
E
TUBULAR TRUSS SPAR

SECTION B-B
(ROTATED 90° CCW)
DEPICTED FOR 40536
UPPER PANEL, LOWER SIMILAR

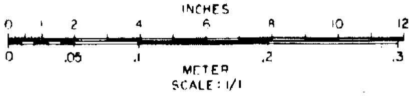

F
F
WELD BOND STRINGERS
& FRAME TO SKIN
BLIND FASTENER
WELD BOND

SECTION D-D

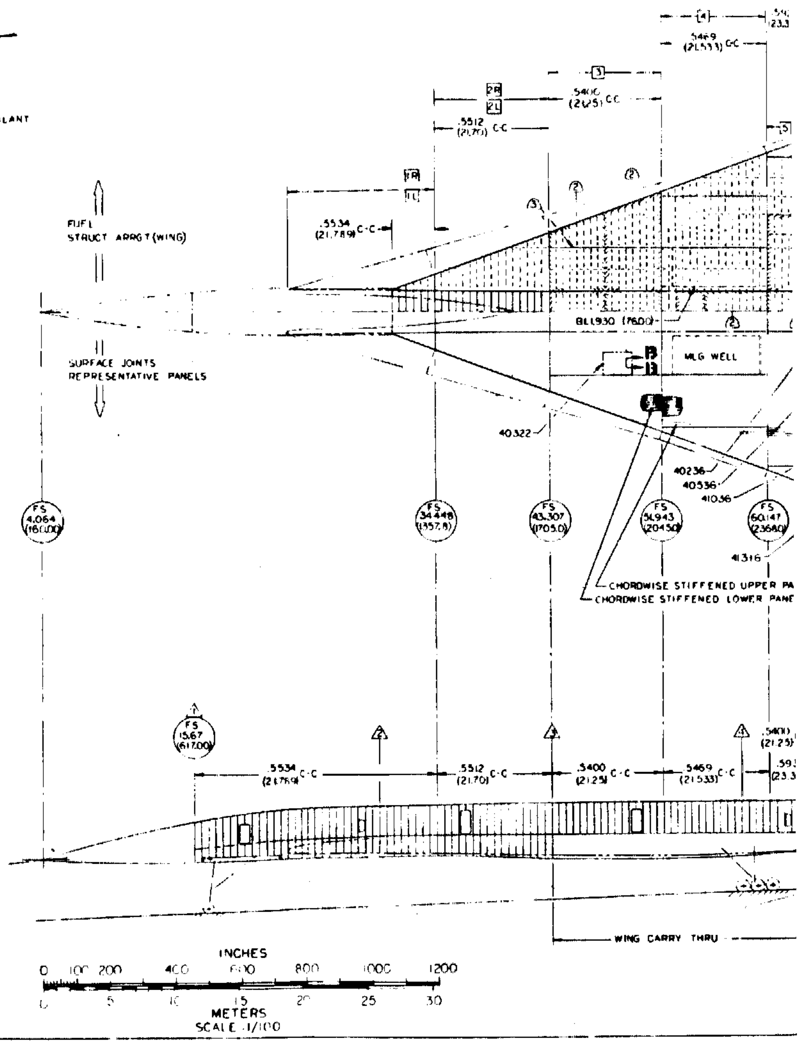


VIEW F-F

TUBULAR TRUSS SPAR								
LOCATION	CAPS				DIAGONALS			
	40322		40536		40322		40536	
	A	B	A	B	C	D	C	D
UPPER	.0038 (.150)	.0381 (1.50)	.0307 (1.21)	.0508 (2.00)	.0006 (.025)	.0381 (1.50)	.00127 (0.05)	.0413 (1.625)
LOWER	.0053 (.210)	.0381 (1.50)	.0358 (1.41)	.0635 (2.50)				


INCHES
0
2
4
6
8
10
12
0
.05
.1
.2
.3
METER

SCALE: 1/1


FUEL
STRUCT ARRGT (WING)
SURFACE JOINTS
REPRESENTATIVE PANELS
1R
1L
2R
2L
3
4
5
.5534
(21.789) C-C
.5512
(21.70) C-C
.5400
(21.25) C-C
.5469
(21.533) C-C
.591
(23.3)
BL1.930 (76.00)
MLG WELL
B
B
40322
40236
40536
41036
41316
FS
4.064
(160.00)
FS
34.448
(1357.8)
FS
43.307
(1705.0)
FS
51.943
(2045.0)
FS
60.147
(2368.0)
FS
15.67
(617.00)
CHORDWISE STIFFENED UPPER PA
CHORDWISE STIFFENED LOWER PANE
.5534
(21.789) C-C
.5512
(21.70) C-C
.5400
(21.25) C-C
.5469
(21.533) C-C
.5400
(21.25)
.591
(23.3)
WING CARRY THRU
INCHES
0
100
200
400
600
800
1000
1200
0
5
10
15
20
25
30
METERS

SCALE 1/100

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FOLDOUT FRAME

FOLDOUT FRAME

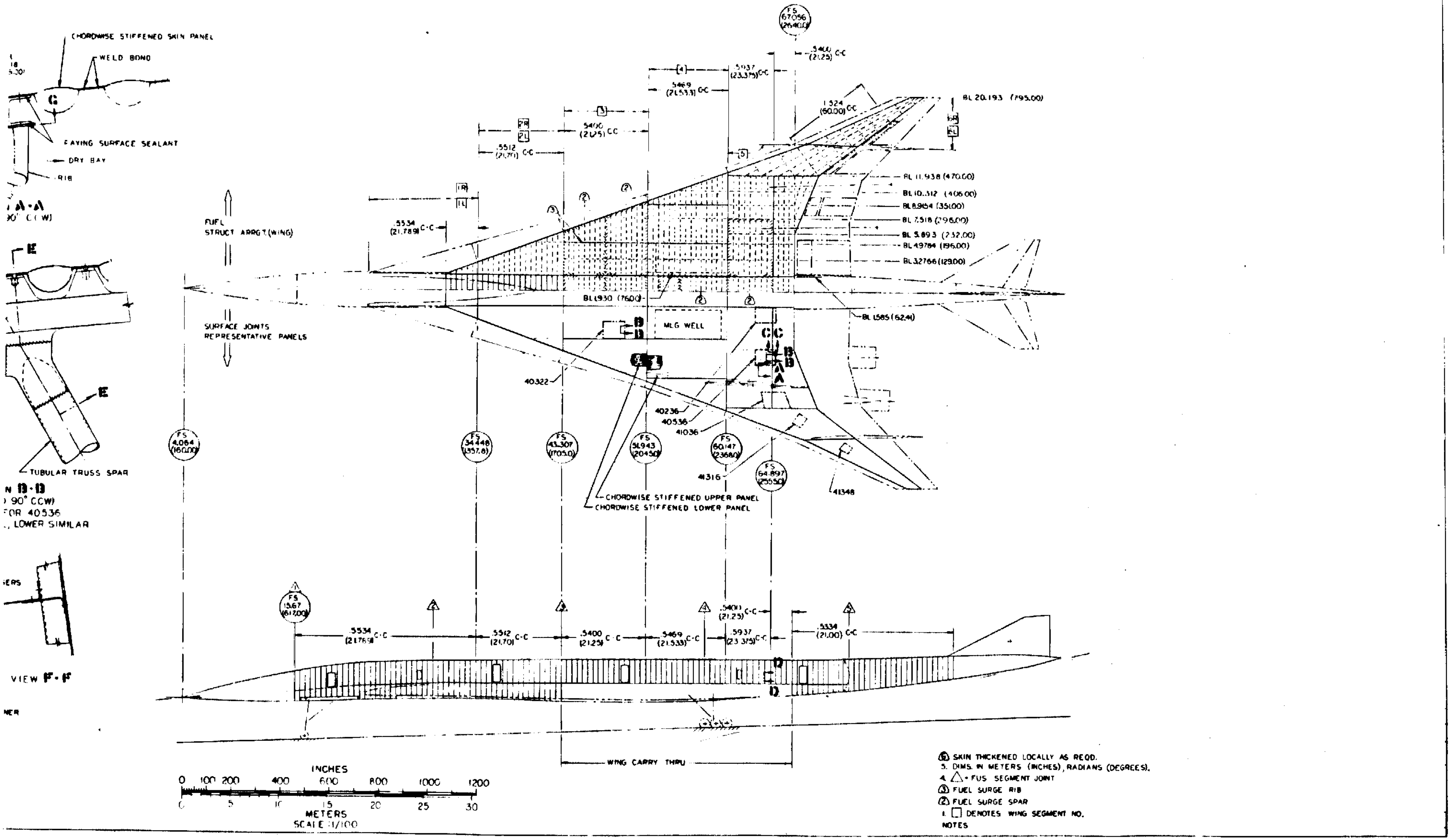
CHORDWISE STIFFENED SKIN PANEL
WELD BOND
FAYING SURFACE SEALANT
DRY BAY
RIB
A-A
90° C CW)
E
TUBULAR TRUSS SPAR
B-B
90° CCW)
FOR 40536
LOWER SIMILAR
VIEW F-F
FUEL STRUCT ARRGT (WING)
SURFACE JOINTS REPRESENTATIVE PANELS
FS 4.064 (160.00)
FS 34.448 (1357.8)
FS 43.307 (1705.0)
FS 51.943 (2045.0)
FS 60.147 (2368.0)
FS 64.897 (2555.0)
FS 67.056 (2640.0)
FS 15.67 (617.00)
.5534 (21.789) C-C
.5512 (21.70) C-C
.5400 (21.25) C-C
.5469 (21.533) C-C
.5937 (23.375) C-C
1.524 (60.00) C-C
.5334 (21.00) C-C
BL 20.193 (795.00)
BL 11.938 (470.00)
BL 10.312 (406.00)
BL 8.9154 (351.00)
BL 7.518 (296.00)
BL 5.893 (232.00)
BL 4.9784 (196.00)
BL 3.2766 (129.00)
BL 1.930 (76.00)
BL 1.585 (62.41)
MLG WELL
C-C
40322
40236
40536
41036
41316
41348
CHORDWISE STIFFENED UPPER PANEL
CHORDWISE STIFFENED LOWER PANEL
D-D
WING CARRY THRU
INCHES
0 100 200 400 600 800 1000 1200
0 5 10 15 20 25 30
METERS
SCALE 1/100
⑥ SKIN THICKENED LOCALLY AS REQD.
5. DIMS. IN METERS (INCHES), RADIANS (DEGREES).
4. △ • FUS SEGMENT JOINT
③ FUEL SURGE RIB
② FUEL SURGE SPAR
1. ☐ DENOTES WING SEGMENT NO.
NOTES

Figure 18-1. Chordwise Stiffened Wing Design - Fasteners

FOLDOUT FRAME 2

FOLDOUT FRAME 3

- Large surface panel assemblies are used.
- The "fastener" joining method is used throughout the entire wing in accordance with the matrix of Table 18-2. As discussed in the Design Approach section, this indicates that major components are joined together with fasteners in order to permit conventional assembly of major components to obtain the aircraft structure. Welding, weld bonding and other joining methods are used, where advantageous, in fabricating individual components such as spars, surfaces and other subassemblies.

The arrangement of the structure is based on the results of the analytical studies (Section 12, Structural Concepts Analysis) and is used to establish the mass estimates reported in Section 15, Mass.

The chordwise-beaded surfaces used in the structure of Figure 18-1 are similar in some respects to those used in the L-2000-7A and the YF-12/SR-71. However, the latter two aircraft used the bead design noted as "corrugation" rather than the circular arc-convex configuration shown in Figure 18-1. The latter bead design has been used because of its structural efficiency and the reduced eccentricity encountered at the terminations of the beads. It is recognized that the circular arc-convex beaded design may result in an increased drag penalty as well as increased danger of damage during aircraft maintenance, hail storms and from foreign object damage (FOD). However, the analytical results of the chordwise stiffened panel designs reported in Section 12, Structural Concepts Analysis, indicated the minimum mass potential for the design shown. The design is directed towards alleviating thermal stresses while still utilizing both skins in resisting shear. Also, heat transfer to the fuel heat sink is reduced, relative to a single skin design, because the fuel does not contact the outer skin in the areas covered by the beads in the inner skin. The cavities enclosed by the beads are vented to ambient atmosphere by means of a hole in the outer skin at each end of a cavity. With the beaded inner and outer skins the surfaces do not participate in resisting wing bending and all bending material is concentrated in the spar caps. This uncoupling of bending and torsional material permits either the bending or torsional stiffness to be varied independently of the other.

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A major manufacturing joint, and a change in direction of the spars, occur at BL 470. This is the location of the outer rib for the outboard engine as well as the joint between the inner wing and the highly swept outer wing and tip.

The "fastener" concept used for assembly major components in the chordwise stiffened wing structure means that fasteners are used for the surface/substructure attachments. Problems arising in this area are much more severe if flush surface/substructure attachments are required.

Sources of attachment problems include those listed below:

- The thin outer skin thickness anticipated over much of the wing makes it difficult to avoid the "feather edges" encountered if a countersink extends through a sheet. It was strongly desired to avoid this condition even though feather edges were accepted on the L-2000-7A and the YF-12/SR-71 aircraft.
- Ti, CRES and similar rivets are difficult to drive and require relatively massive bucking bars and sizable access space. The submerged spar caps shown in Section B-B and the "I" cap spar shown in Section C-C of Figure 18-1 illustrates the restricted access for installing surface attachments.
- Since individual spars are not of fail-safe design, the failure of a spar results in sizable loads transmitted from the surface to adjacent spars and require numerous attachments for load redistribution.
- Upset type fasteners, particularly with thin skin, can result in deformation of the skin and damage or destruction of the bond between the inner and outer skins.

For preliminary design, minimum exterior skin gauges were assumed to be 0.020 inch for the lower surface and 0.015 inch for the upper surface. The minimum gauge of the inner skin was 0.010. Shear type (shallow head) flush head fasteners were considered acceptable in the outer skin.

Studies of access requirements for fastener installation yielded the results shown in Figure 18-2. As shown, blind fasteners require the minimum space. The space requirement for Hi-Tigue fasteners is based on the use of the special tool described

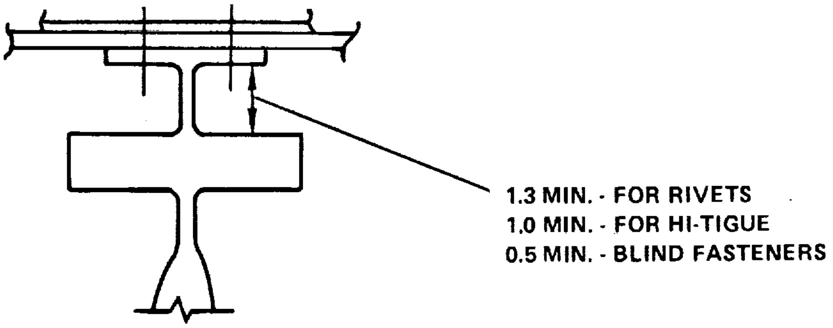


Figure 18-2. Installation Space Requirements for Various Fasteners

in Reference (3). Fastener studies also revealed an interesting new fastener, the "Cherrybuck." This is a one-piece bimetal fastener which has a Commercially Pure (CP) or ductile Ti tail on the end of a Ti-6Al-4V shank. This fastener, used on the B-1 and the F-5E, offers a high strength (95 ksi shear), lower weight than 2 or 3 piece fasteners, and a ductile Ti tail which upsets to 1.5 D without special tools. Investigation also was made of the LS 9714 rivet, a 100-degree miniature head solid rivet designed during the L-2000-7A program. The LS 9714 rivet was developed in two materials: A-286 CRES (95,000 psi min shear) and C.P. titanium (65,000 psi min shear). Comparison shows that the Cherrybuck rivet (all Ti) provides the same shear strength as the LS 9714 A-286 CRES rivet at a lower weight. These limited investigations indicate that:

- Cherrybuck rivets appear useful where an upset flush rivet is acceptable and where the 95 ksi shear strength is advantageous
- LS 9714 CP Ti rivets appear usable when an upset type of flush rivet is satisfactory and the higher shear strength (and probable higher cost) of a Cherrybuck rivet are not required.

As stated previously, concern is felt that upset type fasteners can result in deformation of thin skin and damage or destruction of the bond between the inner and outer skin. Suitable tests are recommended for resolving this question. Where this is found to be a problem, Hi-Lok or Hi-Tigue fasteners are applicable. Hi-Tigue fasteners are indicated where fatigue considerations are significant.

To avoid the feather edge problem in thin skin areas, several approaches have been considered. The two most interesting possibilities are discussed below:

- In one approach, thickened strips would be incorporated in the outer skin sheet in the areas directly above spar (and rib) caps. These strips would be welded in place as part of the process of welding sheets together to obtain the large outer skin weldment required for the large surface panel assemblies used. The thickness of these strips would be at least 0.010 ins. more than the depth of countersinks for flush fasteners. These strips would be machined so that the increased thickness area at fasteners would fair back into the basic skin thickness at a slope of 15/1 or more. This design would result in more welding in the outer skin than would otherwise be required and the thickened areas also would lead to a higher skin weight. Also, surface smoothness may be degraded somewhat.
- In another approach, the outer skins would be chemically milled from sheets whose initial thickness would be at least 0.010 ins. more than the depth of countersinks for flush fasteners. The chem milling would reduce the skin thickness as required in the non-fastened areas while maintaining the thickened areas around fasteners. Machining would be required to smoothly blend out the sharp edges obtained at the edges of chem milled areas. This approach has some merit, but available data indicates that chem milling etches welded areas about 10-percent faster than basic sheet. This is of some concern since the outer skin is a large weldment. However, this problem might be overcome in various ways, one of which is to chem mill individual skin segments before these segments are welded together to obtain the large final external skin required.

In approaching the thin skin/feather edge problem, it appears desirable to consider the steps outlined below.

- Identify those areas where the degraded structural quality associated with feather edges is permissible and accept the feather edge condition in those areas.
- Where possible, investigate smaller and more numerous fasteners as a potential method for avoiding the feather edge problem.

- Where warranted, the "chem milled skin" design outlined above appears feasible and the most practical choice at the present.

Spanwise Stiffened Wing Design — Fasteners

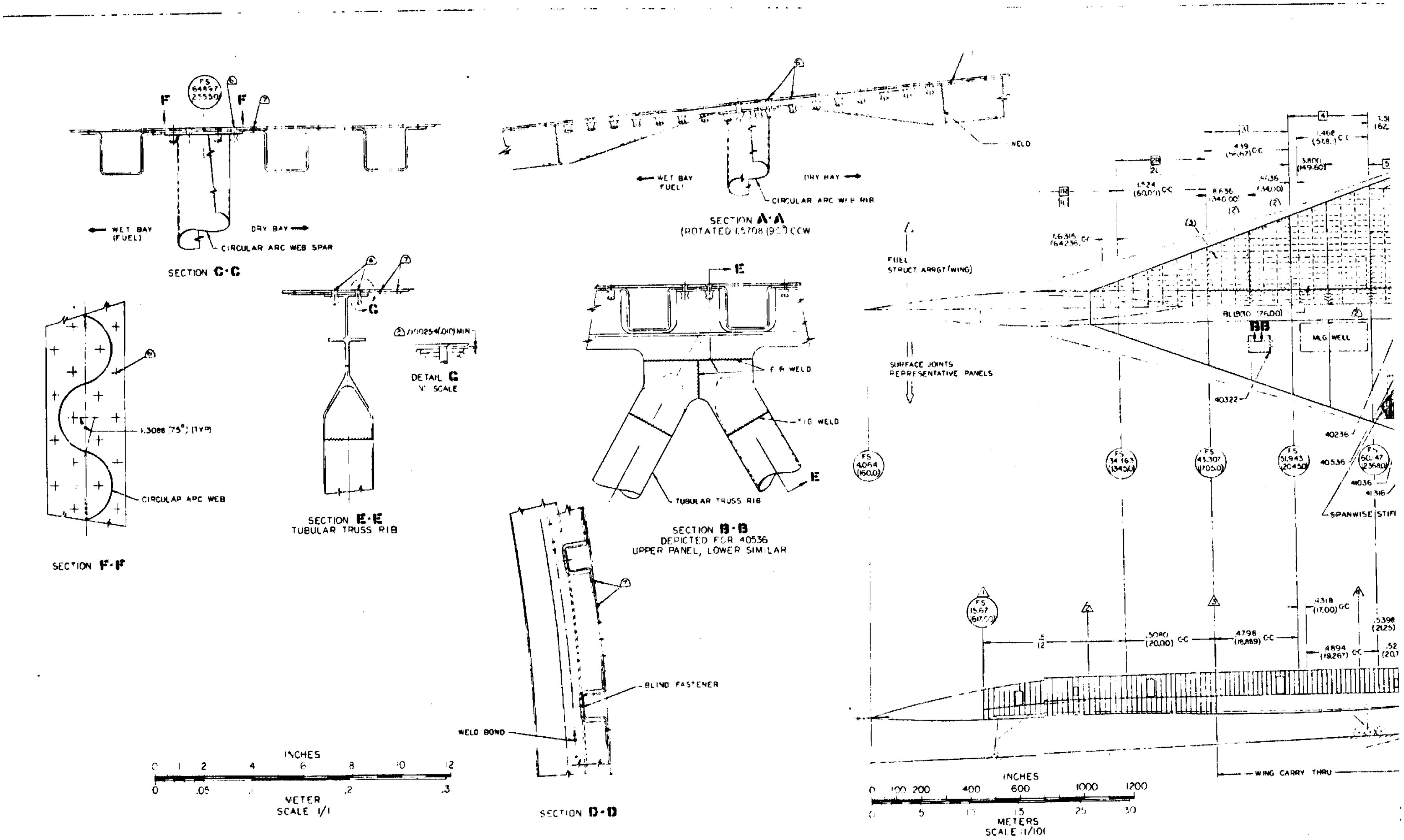
The structural arrangement drawing for the spanwise stiffened wing design is presented in Figure 18-3. The planform view of the airplane shows the arrangement of substructure, fuel tanks and surface panel joints.

General features of the spanwise stiffened structures in Figure 18-3 include those listed below:

- Relatively widely spaced spars with closely spaced ribs.
- Spars and ribs are of truss design except where they serve as fuel tank walls or fuel tank baffles.
- At fuel tank wall and fuel baffle locations, spars and ribs have welded corrugated webs topped by I-section caps mechanically attached to the welded corrugated webs.
- Large surface panel assemblies are used.
- The "fastener" joining method is used throughout the entire wing in accordance with the matrix of Table 18-1. As discussed earlier, this means that major components are joined together with fasteners in order to permit conventional assembly of major components of the aircraft structure. As with all "fastener" designs, welding, weld bonding and other joining methods are used where advantageous in fabricating individual components such as spars and other subassemblies.

The spanwise stiffened wing structures of Figure 18-3 are configured on the basis of the factors noted below:

- Ribs are spaced relatively closely. As shown in the planview of Figure 18-3, the spacing is approximately 20 inches. Major ribs include two per engine, two per MLG strut and a rib at each side of the fuselage. As shown on the figure, major ribs are extended fore and aft, in some cases, to facilitate load distributions, assist in damage tolerance or serve as fuel baffles.


FS 64.897 (2555.0)
F
F
WET BAY (FUEL)
DRY BAY
CIRCULAR ARC WEB SPAR
SECTION C-C
WELD
WET BAY FUEL
DRY BAY
CIRCULAR ARC WEB RIB
SECTION A-A
(ROTATED 1.5708 (90°) CCW
C
.000254 (.010) MIN
DETAIL G
V SCALE
1.3088 (75°) (TYP)
CIRCULAR ARC WEB
SECTION F-F
SECTION E-E
TUBULAR TRUSS RIB
E
E
F.R. WELD
TIG WELD
TUBULAR TRUSS RIB
SECTION B-B
DEPICTED FOR 40536
UPPER PANEL, LOWER SIMILAR
BLIND FASTENER
WELD BOND
SECTION D-D
INCHES
0 1 2 4 6 8 10 12
0 .05 .1 .2 .3
METER
SCALE 1/1
FUEL
STRUCT ARRGT (WING)
SURFACE JOINTS
REPRESENTATIVE PANELS
1.6315 CC (64.236)
1.524 (60.00) CC
8.636 (340.00) (2)
.439 (17.27) CC
4.136 (34.00) (2)
3.800 (149.60)
1.468 (57.8) CC
BL 1.930 (76.00)
BB
MLG WELL
40322
40236
40536
40036
41316
SPANWISE STIFF
FS 4.064 (160.0)
FS 34.163 (1345.0)
FS 43.307 (1705.0)
FS 51.943 (2045.0)
FS 60.47 (2380.0)
FS 15.67 (617.0)
.5080 (20.00) CC
.4798 (18.889) CC
.4318 (17.00) CC
.4894 (19.267) CC
.5398 (21.25)
.52 (20.7
WING CARRY THRU
INCHES
0 100 200 400 600 1000 1200
0 5 10 15 25 30
METERS
SCALE 1/100

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FOLDOUT FRAME 1

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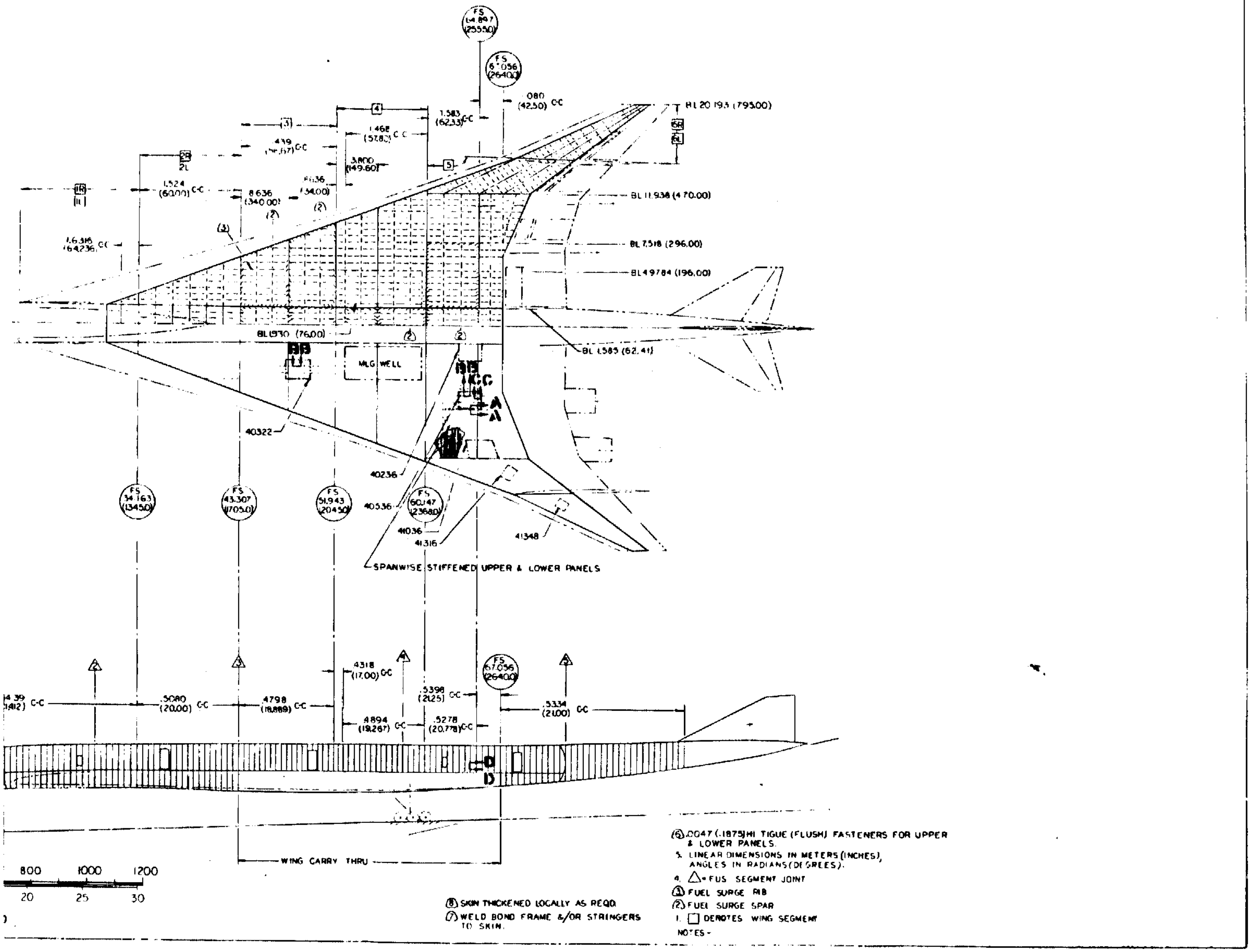

FS 64.897 (2555.0)
FS 67.056 (2640.0)
.080 (42.50) C-C
1.583 (62.33) C-C
1.468 (57.80) C-C
3.800 (149.60)
4
3
5
.439 (56.67) C-C
2R
2L
1R
1L
6R
6L
1.524 (60.00) C-C
8.636 (340.00)
.8636 (34.00)
1.6316 (64.236) C-C
(2)
(3)
BL 20.193 (795.00)
BL 11.938 (470.00)
BL 7.518 (296.00)
BL 4.9784 (196.00)
BL 1.585 (62.41)
BL 1.930 (76.00)
BB
CC
A
MLG WELL
40322
40236
40536
41036
41316
41348
FS 34.163 (1345.0)
FS 43.307 (1705.0)
FS 51.943 (2045.0)
FS 60.147 (2368.0)
SPANWISE STIFFENED UPPER & LOWER PANELS
4
.4318 (17.00) C-C
.5398 (21.25) C-C
.4894 (19.267) C-C
.5278 (20.778) C-C
.5334 (21.00) C-C
.5080 (20.00) C-C
.4798 (18.889) C-C
4.39 (1412) C-C
D
WING CARRY THRU
800
1000
1200
20
25
30
NOTES -
1. ☐ DENOTES WING SEGMENT
(2) FUEL SURGE SPAR
(3) FUEL SURGE RIB
4. △ = FUS SEGMENT JOINT
5. LINEAR DIMENSIONS IN METERS (INCHES), ANGLES IN RADIANS (DEGREES).
(6) .0047 (.1875) HI TIGUE (FLUSH) FASTENERS FOR UPPER & LOWER PANELS.
(7) WELD BOND FRAME &/OR STRINGERS TO SKIN.
(8) SKIN THICKENED LOCALLY AS REQD.

Figure 18-3. Spanwise Stiffened Wing Design - Fasteners

FOLDOUT FRAME 3

- A dry bay completely surrounds the MLG well so that no MLG support structure attaches directly to structure serving as a fuel tank wall. This reduces the possibility of damage to the fuel tanks in the event of a hard landing or similar occurrence.
- Forward of FS 1705, the forward end of center section fuel, spars do not extend through the fuselage and the wing is supported from the side of the fuselage.
- The wing is built in segments which, except for the aft main box segment, include all wing structure from one leading edge to the opposite leading edge. The aft main wing box extends from the MLG cutout to the rear beam and from one fixed fin to the opposite fixed fin (a span of 100 ft.). This manufacturing breakdown is developed from the structural arrangement and is based on minimizing the number of segments while avoiding excessive size and difficult assembly and handling of the various segments.

The sheet metal hat stiffened surface panel design resulting from the analytical design study is shown. The surface assembly, which consists of the outer skin and hat stiffeners, is attached to the rib assembly comprised of the rib chords and diagonals fabricated as one wledment. See Section B-B of Figure 18-3.

With the hat-stiffened surface (closed section stiffeners), it is noted that heat transfer from the slipstream to the fuel heat sink is reduced in value when compared to the single skin-open section concepts, since more of the outside skin is in contact with the fuel.

Weld bonding has been selected for the stringer/skin attachment after consideration of several alternate methods. Surface panel sizes are shown on the planform view of Figure 18-3. For forming and fabricating individual panels, a size limit of 15 by 35 feet has been used. This limit is discussed in Reference (3). Where a joining operation involves bonding only, the panel length limit has been extended to 50 feet. This special length limit has been exploited wherever possible.

In the aft main wing box area the surface length extends from BL 62.41L to BL 470L and from BL 62.41R to BL 470R. A splice of the hat stiffeners, is integrated into the design (Section A-A of Figure 18-3) to accommodate a fuel rib at BL 296L and R.

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This surface configuration eliminates a chordwise joint at BL 296L and R, however, results in a "fastened" (rather than "fastened/bonded") surface splice at BL 62.41L and R.

In fuel stowage areas, access is required to the interior of the tanks. For this purpose, two access doors in the surface are provided for each individual tank or isolated portion thereof. The design intent is to minimize the access doors in the outer surface by exploiting the openings between diagonals in truss type ribs and spars was emphasized. Access door cutouts in the surface result in more weight penalty in the spanwise than chordwise stiffened design since the surfaces contain bending as well as shear material.

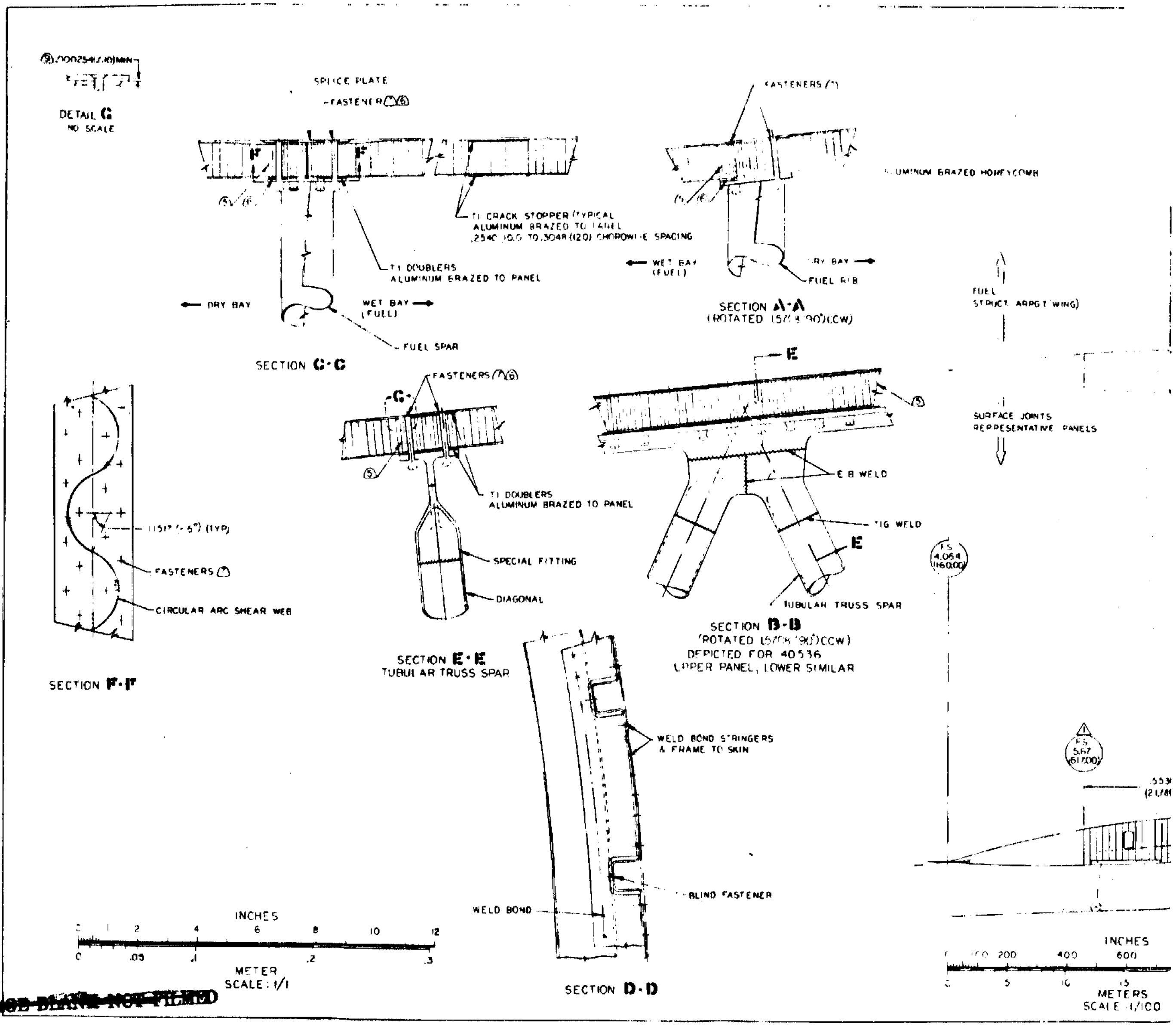
Additional provisions for fail-safe and damage tolerance were not required for the spanwise stiffened design as indicated in Section 13, Fatigue and Fail-Safe.

The "Fastener" concept used for assembling major components in the spanwise stiffened wing structures of Figure 18-3 means that fasteners are used for the surface/substructure attachments.

A major manufacturing joint and change in direction of the spars and surface panels occur at BL 470. This is the location of the outer rib for the outboard engine as well as the joint between the inner and outer wing.

Monocoque Wing Design - Fastener

The structural arrangement drawing of the monocoque wing design is presented in Figure 18-4. The surface panels consist of aluminum brazed honeycomb core sandwich panels with densified core for the "fastener" approach. The arrangement of the substructure, fuel tanks and surface panel joints is indicated on the wing planform.


⑤ .000254(.010)MIN
DETAIL G
NO SCALE
SPLICE PLATE
FASTENER ⑦⑥
F
F
⑤ ⑥
TI CRACK STOPPER (TYPICAL
ALUMINUM BRAZED TO PANEL
.2540 (0.0 TO .3048 (120) CHORDWISE SPACING
TI DOUBLERS
ALUMINUM BRAZED TO PANEL
DRY BAY
WET BAY
(FUEL)
FUEL SPAR
SECTION C-C
FASTENERS ⑦
⑤ ⑥
ALUMINUM BRAZED HONEYCOMB
WET BAY
(FUEL)
DRY BAY
FUEL RIB
SECTION A-A
(ROTATED 15°(4.90°)CCW)
FUEL
STRUCT. ARRGT WING
SURFACE JOINTS
REPRESENTATIVE PANELS
E
⑤
E B WELD
TIG WELD
E
TUBULAR TRUSS SPAR
FS
4.064
(160.00)
SECTION B-B
(ROTATED 15°(8.90°)CCW)
DEPICTED FOR 40536
UPPER PANEL, LOWER SIMILAR
FASTENERS ⑦⑥
C
⑤
TI DOUBLERS
ALUMINUM BRAZED TO PANEL
SPECIAL FITTING
DIAGONAL
SECTION E-E
TUBULAR TRUSS SPAR
.1517 (.5°) (TYP)
FASTENERS ⑦
CIRCULAR ARC SHEAR WEB
SECTION F-F
WELD BOND STRINGERS
& FRAME TO SKIN
BLIND FASTENER
WELD BOND
SECTION D-D
FS
567
(61.700)
553
(21.78)
INCHES
0 2 4 6 8 10 12
0 .05 .1 .2 .3
METER
SCALE: 1/1
INCHES
0 100 200 400 600
0 5 10 15
METERS
SCALE 1/100

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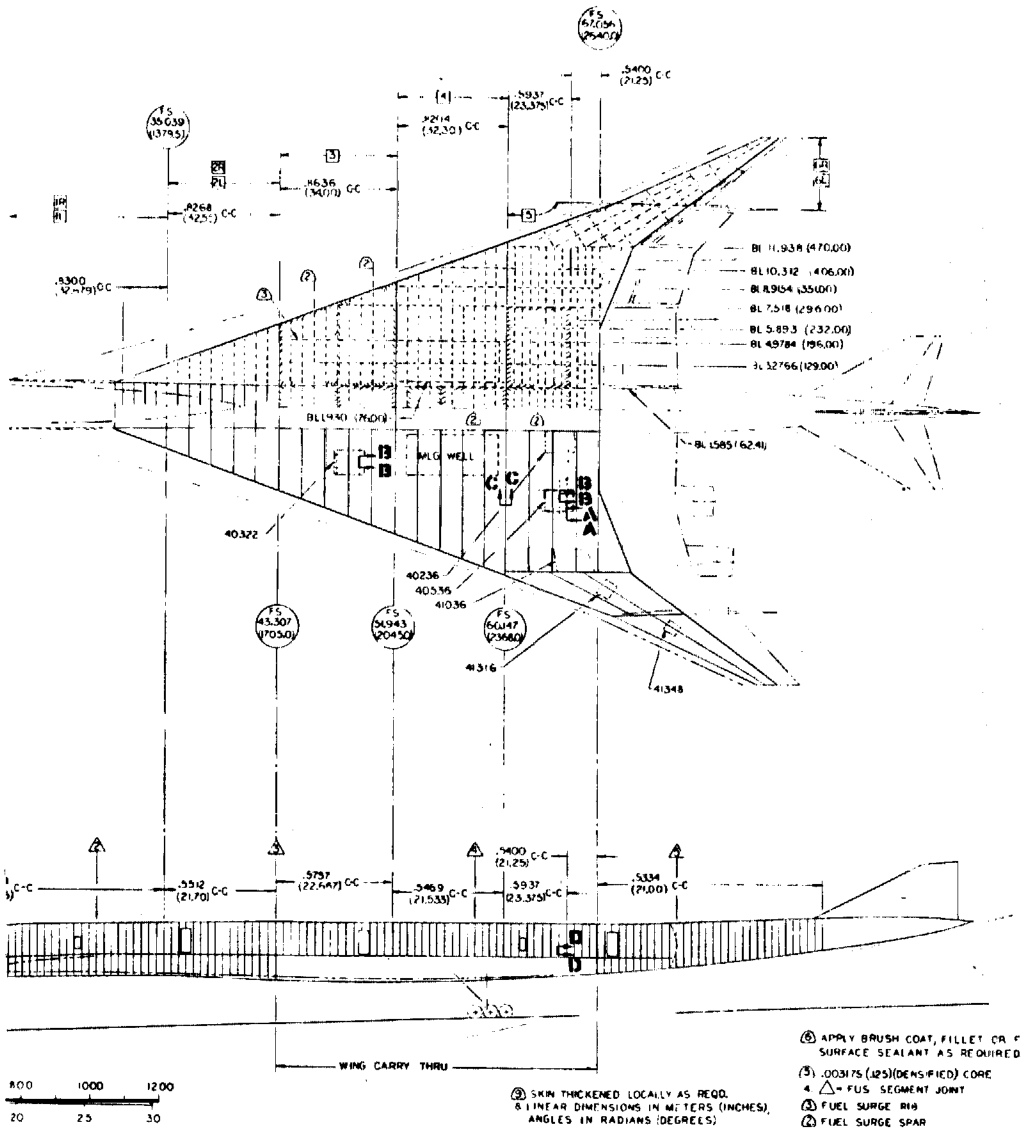
F.S. 35.039 (1379.5)
F.S. 67.056 (2640.0)
1R
1L
2R
2L
3
4
5
6R
6L
.8300 (32.679) C-C
.8268 (42.5?) C-C
.8636 (34.00) C-C
.9214 (32.30) C-C
.5937 (23.375) C-C
.5400 (21.25) C-C
BL 11.938 (470.00)
BL 10.312 (406.00)
BL 8.9154 (351.00)
BL 7.518 (296.00)
BL 5.893 (232.00)
BL 4.9784 (196.00)
BL 3.2766 (129.00)
BL 1.585 (62.41)
BL 1.930 (76.00)
MLG WELL
A
B
C
D
40322
40236
40536
41036
41316
41348
F.S. 43.307 (1705.0)
F.S. 51.943 (2045.0)
F.S. 60.147 (2368.0)
.5512 (21.70) C-C
.5757 (22.667) C-C
.5469 (21.533) C-C
.5400 (21.25) C-C
.5937 (23.375) C-C
.5334 (21.00) C-C
WING CARRY THRU
800
1000
1200
20
25
30

NOTES-

1. ☐ DENOTES WING SEGMENT NO.

② FUEL SURGE SPAR

③ FUEL SURGE RIB

4. △ = FUS SEGMENT JOINT

⑤ .003175 (.125)(DENSIFIED) CORE

⑥ APPLY BRUSH COAT, FILLET OR FAYING SURFACE SEALANT AS REQUIRED.

⑦ .004763(.1875) HI-TIGUE FOR UPPER & LOWER PANELS.

8. LINEAR DIMENSIONS IN METERS (INCHES), ANGLES IN RADIANS (DEGREES)

⑨ SKIN THICKENED LOCALLY AS REQD.

Figure 18-4. Monocoque Wing Design - Fasteners

FOLDOUT FRAME 5

The panel concept selection was based on the results of the analytical design of both the honeycomb and truss core sandwich designs discussed in Section 1, Structural Design Concepts. These panel concept designs were selected after a comprehensive review of numerous candidates and their applicability to a near-term arrow-wing configuration supersonic transport. The types of surface panel studied include:

- Aluminum brazed honeycomb sandwich surfaces made by the "Aeronca" process.
- Adhesive bonded honeycomb sandwich surfaces.
- Welded honeycomb sandwich (STRESSKIN) surfaces.
- Trusscore sandwich surfaces, a design using a unidirectionally corrugated core sheet between the sandwich face skins.

In addition to the above, both the Rohr Liquid Interface Diffusion (LID) and Northrop Nor-Ti-Bond processes were considered and evaluated for the fabrication of brazed sandwich surface panels. In the Nor-Ti-Bond process, copper is electroplated on the edges of the honeycomb cell walls before the core is brazed to the face sheets. The brazing step results in a diffusion brazed joint between the core and face sheets. The Rohr LID process is similar except that several elements, rather than copper alone, are electroplated on the edges of the honeycomb core cells. Both processes provide good mechanical properties as well as operating temperature capabilities. However, both have demanding tolerance requirements. For example, the core/face sheet gap must not exceed 0.005 ins. In contrast, the "Aeronca" process aluminum braze process can tolerate a core/face sheet gap up to 0.002 ins. Because of the manufacturing difficulties foreseen in applying the LID and Nor-Ti-Bond processes to the large wing surfaces for the near-term airplane, detail drawings were not made.

The waffle type of monocoque surface was not included for application to the large wing surface panels under consideration because of the difficulties and costs foreseen in fabricating and forming such panels.

The surface panels selected for design application use the "Aeronca" aluminum brazing process originally developed by Aeronca for the Boeing SST. In this process, a thin aluminum foil sheet (3003 alloy) is placed between the core and face sheet, or

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other surfaces to be joined, and the brazing step conducted at a temperature of 1240 F. A detailed discussion of the Aeronca development work of this fabrication method is described in Reference 2.

The Aeronca process results in a thin coating of aluminum over the entire interior surface of the sandwich. This includes the inner surfaces of the face sheets as well as the walls of the honeycomb cells. As can be seen, this indicates that a significant amount of parasitic weight of unnecessary aluminum braze alloy is incorporated in the final brazed sandwich. This aluminum coating also results in a relatively high heat transfer to the fuel and substructure. When non-perforated core is used, each cell is hermetically sealed after brazing. This feature prevents fuel or other material from entering adjacent cells if the fuel or other liquid should somehow enter the cell(s) in a particular area.

The Aeronca aluminum braze process, which is fully discussed in Reference (4), has a number of advantageous features relative to design. These features include:

- The core depth can be varied, or tapered, within a given brazed panel.
- Separate doublers (or fail safe straps) can be incorporated within a brazed panel.
- An appreciable gap can be tolerated between the core and face sheets. Successful brazing has been accomplished with localized gaps up to 0.020 inches, which indicates a realistic process for application to large panels (such as wing surfaces).
- The core density and cell size can be varied within a brazed panel if required.
- Hermetic sealing of each cell, obtained with non-perforated core, means that fuel (or other liquid) which penetrates one cell is not automatically free to enter all other cells in the sandwich.
- No confining restraints apply relative to the thicknesses of the face sheets, the core and inserts. The thickness variation among these elements can be very large and is not restricted as is the case when, for example, the core is spotwelded to the face sheet.

- Considerable development effort has been expended on this type of structure and an appreciable amount of data is available. Boeing has received considerable DOT funding for development and tests of this structural concept.

The Aeronca process does, however, have a few disadvantages. Some examples include:

- A fairly significant weight penalty is incurred by the fact that 3003 aluminum braze alloy coats the entire interior of the sandwich. "Stop-off" and other techniques to alleviate this situation are discussed in Reference (4).
- The aluminum coating on the interior surfaces of the sandwich results in relatively high heat transfer to the fuel and substructure.
- The fact that the braze alloy melts and flows causes difficulties in brazing joints which have a significant slope (≥about 20-degrees) relative to the basic plane of the panel. Aeronca has claimed development of proprietary techniques for alleviating this condition.
- Repair techniques need further development. Some work has been done using bonding techniques which appear promising. Welded repairs are not feasible in close proximity to aluminum brazed joints.
- Fabrication of large panels, postulated as feasible for this study, will require careful control of brazing temperature, dimensional changes and other factors.

For this study, a maximum brazed panel size of 68 inches by 40 feet has been postulated. This size limit, based on discussions with Aeronca personnel, contrasts with a size of approximately 3 feet x 25 feet for the largest panel brazed to date. At the beginning of this study, no data was available to confirm the fabricability of thick panels with heavy skins. However, data from Reference 2 indicates the feasibility of sandwich thicknesses to 1.60 inches and face sheets to 0.156 inches.

The general features of the monocoque design are enumerated below:

- Numerous spanwise spars which are of truss design except where a spar serves as a fuel tank wall or fuel surge pressure spar.

- At fuel tank wall locations, spars have welded circular arc webs.
- Relatively few chordwise ribs are used. These also are of the truss type except where the rib serves as a tank wall. In such applications, welded corrugated web ribs are used.
- The "fastener" joining method is used throughout the wing in accordance with the matrix of Table 18-2. As discussed in the Design Approach section, this indicates that major components are joined together with fasteners in order to permit conventional assembly of major components. Welding and other joining methods are used, where advantageous, in fabricating individual components such as spars and other subassemblies.
- Surface panels are long and narrow, consistent with the postulated size limit of 68 inches by 40 feet. With the spar spacing used, surface splices are located per Figure 18-4.

Two types of wing surface joints are utilized in the inner wing. A "balanced" (double shear) joint is proposed for heavily loaded areas (Section C-C of Figure 18-4) and where advantageous in structural mass. Section C-C illustrates such a joint at a flush surface splice. In general, all flush fasteners are envisioned as having shallow ("shear") heads to permit minimum countersink depths. The splice plate thickness is then made at least 0.010 inches greater than the fastener head height to avoid "feather edges" and the resulting degradation in structural quality. Note that Section C-C specifies densified (1/8 inch cell) core at the fasteners. In tests conducted by Aeronca, the 1/8 inch cell proved adequate in strength to permit full torquing of a 1/4 inch diameter fastener (to 175 in.-lb. torque) as well as the elimination of separate panel closure. A locally thickened pad machined integral with the skin is shown for load transfer from the face sheet to the splice plate. This approach requires the use of a suitably thick sheet or plate. An alternate approach is welding thicker strips into the face sheet at panel edges and other locations requireing additional thickness.

In the wing apex and other lightly-loaded areas, the "single-edge" panel splice is used with a zee-closure at the edge of the panel.

As previously noted, core cell size can be varied within a brazed panel. However, to date this has been done by poke welding (resistance welding) individual cell walls together at the interface joint between the different cell sizes. This poke welding, done with vary small electrodes, is laborious and expensive. Development is needed for a procedure involving a brazed joint at the interface joint between the two different core cell size areas. Another potential solution is the use of alternate core cell shapes. This is an attempt to reduce the core cell joint areas and weights and to lower the heat transfer rate through a brazed panel.

The aluminum brazed panel design is adaptable to the incorporation of fail-safe straps. As shown a certain proportion of the face sheet material is shown segregated and incorporated in failsafe straps brazed to the face sheet during panel fabrication. It is assumed that these brazed straps will be effective as crack-stoppers. However, tests are required to verify that these integrally brazed straps will be effective crack stoppers.

The substructure in the inboard wing is similar to that used in the chordwise and spanwise stiffened wing designs discussed previously. This indicates that:

- Spars and ribs are of welded truss design except where used as tank walls or fuel baffles. This approach is used to provide good internal access for manufacturing and structural inspection purposes.
- At tank walls, spars and ribs have welded circular arc webs. The web cap directly contacts the surface sandwich as shown in Section A-A of Figure 18-4. At rib/spar joints, a splice plate is used between the web cap and the surface for load transfer. As previously noted, Sciaky has verified the practicality of welding the caps to the circular arc webs at angles as much as 15-degrees from normal to the web.

Hi-Lok or Hi-Tigue fasteners are envisioned for surface/substructure attachments in tank areas.

Feather edges at countersinks, and the associated degraded structural quality, are avoided wherever undesirable by local thickening of the outer skin. Thickness variation in the outer skins is envisioned as obtained by chemical milling/machining.

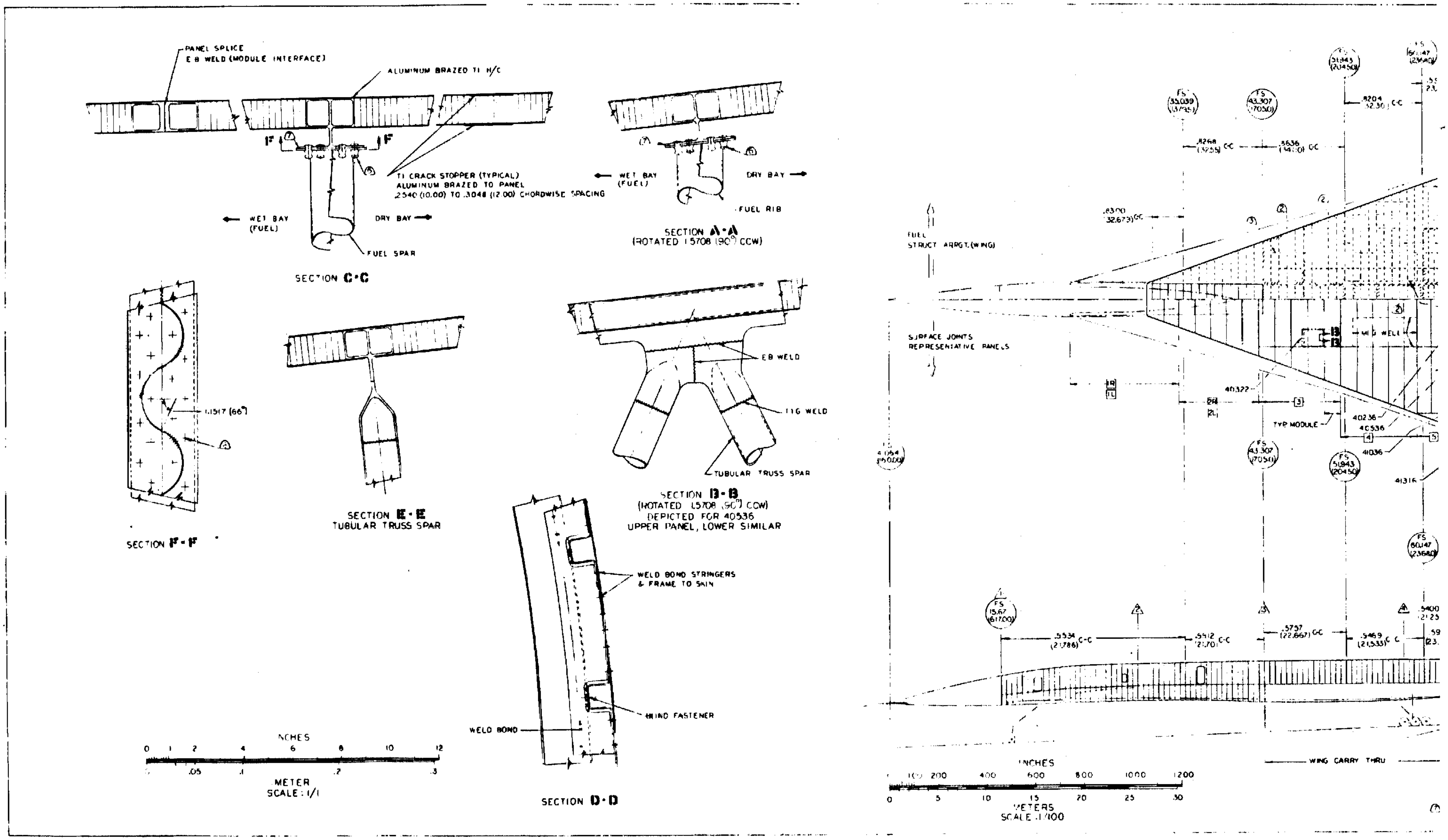
The upper and lower surfaces of the outer wing and wing tip are composed of separate brazed panels which are fastened to substructure. All or some panels of the upper surface are attached with screws and are removable for inspection and maintenance purposes.

The above approach is based on the desire to avoid numerous access door cutouts in the surfaces and the need for blind fasteners.

Monocoque Wing Design - Welded

The structural arrangement drawing of the monocoque wing design - "welded" approach is shown on Figure 18-5. The surface panels consist of aluminum brazed honeycomb cone sandwich panels with tubular inserts to facilitate welding. The fuel and substructrue arrangements are presented on the wing planform along with the surface panel joints of the representative panels.

The welded structure concept described in this section has many similarities to the fastened structure covered previously. This discussion is primarily intended to define differences in the structural design arising from the use of the welding concept.



PANEL SPLICE
E B WELD (MODULE INTERFACE)
ALUMINUM BRAZED TI H/C
F
F
TI CRACK STOPPER (TYPICAL)
ALUMINUM BRAZED TO PANEL
.2540 (10.00) TO .3048 (12.00) CHORDWISE SPACING
WET BAY (FUEL)
DRY BAY
FUEL SPAR
SECTION C-C
WET BAY (FUEL)
DRY BAY
FUEL RIB
SECTION A-A
(ROTATED 1.5708 (90°) CCW)
1.1517 (66°)
SECTION F-F
SECTION E-E
TUBULAR TRUSS SPAR
EB WELD
TIG WELD
TUBULAR TRUSS SPAR
SECTION B-B
(ROTATED 1.5708 (90°) CCW)
DEPICTED FOR 40536
UPPER PANEL, LOWER SIMILAR
WELD BOND STRINGERS & FRAME TO SKIN
BLIND FASTENER
WELD BOND
SECTION D-D
INCHES
0 1 2 4 6 8 10 12
0 .05 .1 .2 .3
METER
SCALE: 1/1
FUEL
STRUCT ARRGT. (WING)
SURFACE JOINTS
REPRESENTATIVE PANELS
FS 4.064 (160.00)
FS 35.039 (13795)
FS 43.307 (17050)
FS 51.943 (20450)
FS 60.147 (23680)
.8300 (32.673) C-C
.8268 (32.55) C-C
.8636 (34.00) C-C
.8204 (32.30) C-C
40322
1R
1L
2R
2L
3
4
5
TYP MODULE
40236
40536
41036
41316
WING WELL
B
B
FS 15.67 (617.00)
.5534 (21.786) C-C
.5512 (21.70) C-C
.5757 (22.667) C-C
.5469 (21.533) C-C
.5400 (21.25)
WING CARRY THRU
INCHES
0 100 200 400 600 800 1000 1200
0 5 10 15 20 25 30
METERS
SCALE 1/100

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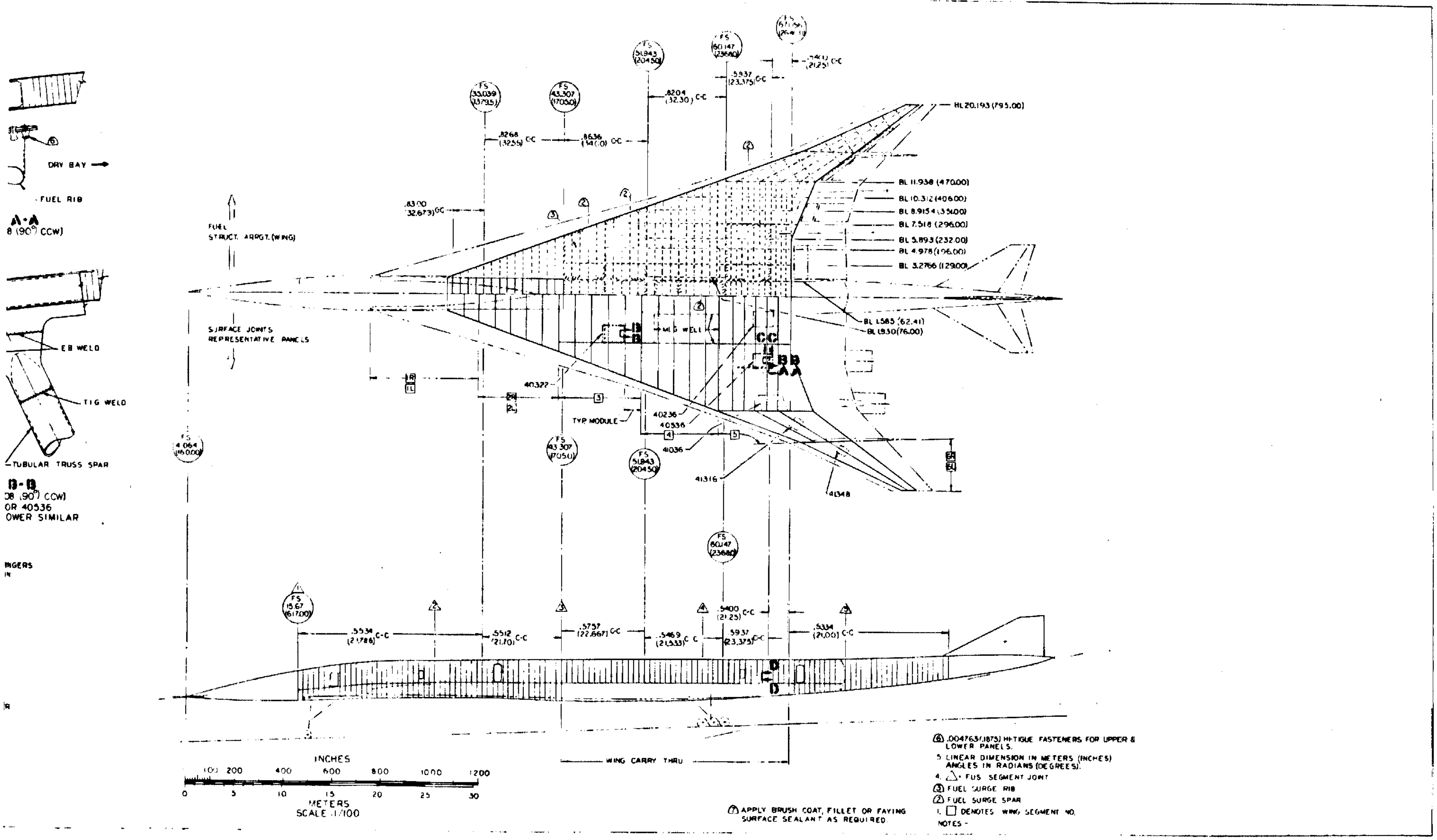
DRY BAY
FUEL RIB
A-A
8 (90° CCW)
EB WELD
TIG WELD
TUBULAR TRUSS SPAR
B-B
08 (90° CCW)
OR 40536
OWER SIMILAR
INGERS
FUEL
STRUCT. ARRGT. (WING)
SURFACE JOINTS
REPRESENTATIVE PANELS
FS 4.064 (160.00)
FS 15.67 (617.00)
FS 33.039 (1379.5)
FS 43.307 (1705.0)
FS 51.943 (2045.0)
FS 60.147 (2368.0)
.8300 (32.673) C-C
.8268 (32.55) C-C
.8636 (34.00) C-C
.8204 (32.30) C-C
.5937 (23.375) C-C
.5400 (21.25) C-C
BL 20.193 (795.00)
BL 11.938 (470.00)
BL 10.312 (406.00)
BL 8.9154 (351.00)
BL 7.518 (296.00)
BL 5.893 (232.00)
BL 4.978 (196.00)
BL 3.2766 (129.00)
BL 1.585 (62.41)
BL 1.930 (76.00)
MLG WELL
TYP MODULE
40322
40236
40536
41036
41316
41348
.5534 (21.788) C-C
.5512 (21.70) C-C
.5757 (22.667) C-C
.5469 (21.533) C-C
.5937 (23.375) C-C
.5334 (21.00) C-C
WING CARRY THRU
INCHES
(0) 200 400 600 800 1000 1200
0 5 10 15 20 25 30
METERS
SCALE 1/100
⑦ APPLY BRUSH COAT, FILLET OR FAYING SURFACE SEALANT AS REQUIRED.
⑥ .004763 (.1875) HI-TIGUE FASTENERS FOR UPPER & LOWER PANELS.
5 LINEAR DIMENSION IN METERS (INCHES) ANGLES IN RADIANS (DEGREES).
4. △ FUS SEGMENT JOINT
③ FUEL SURGE RIB
② FUEL SURGE SPAR
1. ☐ DENOTES WING SEGMENT NO.
NOTES -

Figure 18-5. Monocoque Wing Design - Welded

FOLDOUT FRAME 2

FRAME 3

The design objective in this investigation was development of a completely, or nearly completely, welded structure in an attempt to obtain such anticipated advantages as:

- The elimination of numerous fasteners and the minimization of the number of joints.
- A much simpler problem in sealing fuel tanks. In addition to easier tank sealing, the welding concept permits the avoidance of a significant weight increment associated with tank sealing compounds.

Figure 18-5 depicts the welded structure design developed for the area inboard of BL 470. It employs a unique "module" concept which is illustrated in the sketch of Figure 18-6. As shown therein, a module consists of two adjacent spars, rib segments and an upper and lower surface. The surface panels, twice as wide as the spar spacing, are joined to the adjacent module by spanwise EB welds assembling modules into wing segments. The location of segment joints is shown in Figure 18-5.

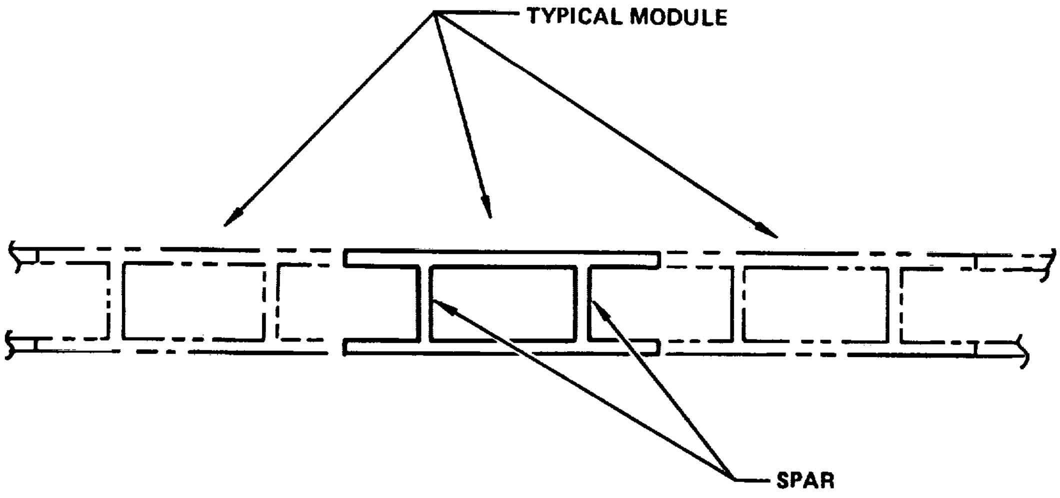

Figure 18-6. Typical Module for Welded Sandwich Wing Structure

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Design guidelines used in developing the welded structure concept include:

- All welding is designed for automation wherever possible.
- EB welding is used wherever feasible, and not for surface joints alone, because of good mechanical properties, a minimum heat-affected zone and a high welding speed.
- All welds are dressed on all surfaces unless impossible.
- Wherever possible, welds are inspectable visually.
- The maximum possible joining of parts is accomplished during the panel brazing process. This is an attempt to minimize assembly labor.

In line with the latter point, spar caps, rib caps, panel closures and other elements are incorporated within the panel brazement. This is illustrated in the sectional views shown in Figure 18-5. This approach provides increased structural depth for spars and ribs. Some panel inserts are extended to the external surface so to provide the capability of extended visual inspection.

In fabrication, the spar and rib caps and panel edge members in a given panel are first EB welded together in a grid. The honeycomb core is poke welded to the inserts and the sandwich skins are then brazed to the insert grid and core to obtain a completed panel brazement. The inserts are shaped to facilitate poke welding to the core. Note that the brazing, performed at approximately 1240 F, stress relieves the welded grid assembly. Since very close tolerances are required for the EB welded joints between modules, the inserts at the module interfaces are made heavier to allow machining on assembly. This is shown in Section C-C of Figure 18-5.

Spars and ribs are of truss design, as with other structural designs in this report, except where they serve as tank walls or fuel baffles. A tank wall spar is shown in Section A-A on Figure 18-5.

Surface panel joints are the balanced, or double edge, type for good structural efficiency and are located midway between spars to allow access for welding equipment and to permit dressing of welds after the joining of modules. Surface panel size is assumed limited to 68 inches in width and 40 feet in length.

Welded joints must be configured and located carefully to ensure that the welding does not damage the brazed joints. The weld/braze clearances shown are preliminary estimates and would require verification tests before finalization.

In the cabin floor area, seat tracks are incorporated in the brazed panels to provide increased structural depth and to facilitate a longer span between supports. The planview of Figure 18-5 shows that both the upper and lower surfaces of the outer wing and wing tip are composed of separate brazed panels. All or some panels of the upper surface are attached with screws and are removable for inspection and maintenance purposes. The major difference from the "Fastened" approach of Figure 18-4 is that the fasteners attaching the lower spar caps to the lower surface have been eliminated in the welded structures of Figure 18-5. This large reduction in the number of fasteners through the lower surface in the welded structure design has been accomplished by incorporating the lower spar caps in the lower surface.

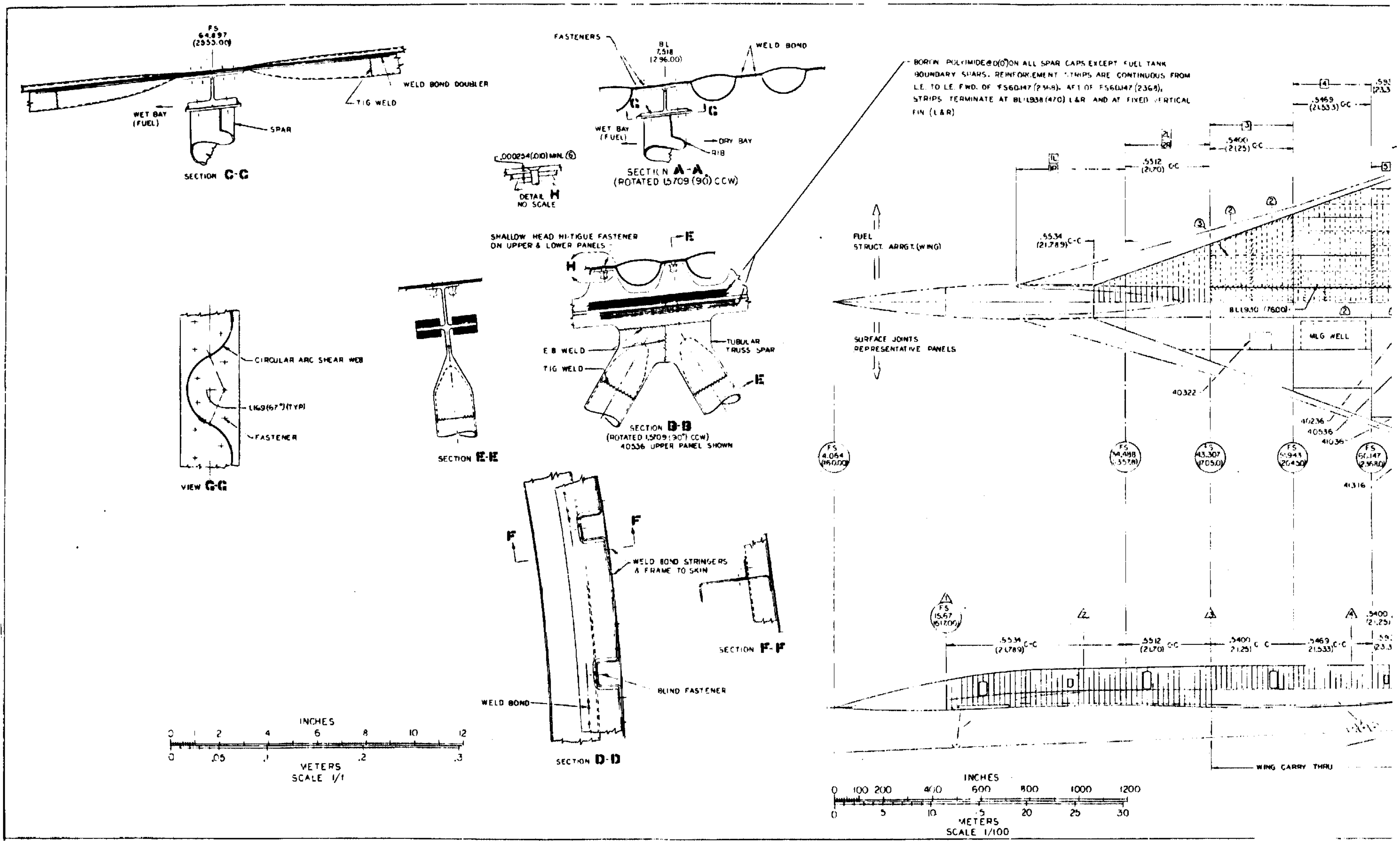
Composite Reinforced Wing Design — Fasteners

The structural arrangement of the composite reinforced wing design is presented in Figure 18-7. The basic arrangement of the structure and design features are identical to the chordwise stiffened wing design of Figure 18-1 and discussed in detail in the chordwise stiffened wing design section.

The structurally efficient circular arc-convex beaded surface panels of titanium alloy (Ti-6Al-4V annealed) are used. The titanium alloy spar caps are reinforced with boron/poyimime (B/PI) as shown in Section E-E of Figure 18-7. The reinforcement strips are continuous from leading edge to leading edge forward of F.S 2368. Aft of F.S. 2368 the reinforcing is continuous from BL 470L to BL 470R.

Skin-Stringer and Frame Fuselage Design — Fasteners

The fuselage structural arrangement shown with the various wing concept designs in Figures 18-1, 18-3, 18-4, 18-5 and 18-7 is a skin-stringer and frame construction. The closed hat stiffener configuration is used in the major portion of the fuselage with the zee stiffener used in the more lightly loaded but pressure critical forebody. The frames are zee-sections with continuous shear clips as indicated on the


FS 64.897 (2555.00)
WELD BOND DOUBLER
TIG WELD
WET BAY (FUEL)
SPAR
SECTION C-C
FASTENERS
BL 7.518 (296.00)
WELD BOND
G
G
WET BAY (FUEL)
DRY BAY
RIB
SECTION A-A (ROTATED 1.5709 (90) CCW)
.000254(.010) MIN. ⑥
DETAIL H NO SCALE
SHALLOW HEAD HI-TIGUE FASTENER ON UPPER & LOWER PANELS
H
E
E
E.B WELD
TIG WELD
TUBULAR TRUSS SPAR
SECTION D-D (ROTATED 1.5709 (90°) CCW) 40536 UPPER PANEL SHOWN
CIRCULAR ARC SHEAR WEB
1.169(67°)(TYP)
FASTENER
VIEW G-G
SECTION E-E
F
F
WELD BOND STRINGERS & FRAME TO SKIN
SECTION F-F
BLIND FASTENER
WELD BOND
SECTION D-D
INCHES
0 2 4 6 8 10 12
0 .05 .1 .2 .3
METERS SCALE 1/1
BORON POLYIMIDE@0(0°)ON ALL SPAR CAPS EXCEPT FUEL TANK BOUNDARY SPARS. REINFORCEMENT STRIPS ARE CONTINUOUS FROM L.E. TO LE. FWD. OF FS60.147 (2368). AFT OF FS60.147 (2368), STRIPS TERMINATE AT BL 11.938 (470) L&R AND AT FIXED VERTICAL FIN (L&R)
1L
1R
2L
2R
3
4
5
.5534 (21.789) C-C
.5512 (21.70) C-C
.5400 (21.25) C-C
.5469 (21.533) C-C
.593 (23.3
FUEL
STRUCT. ARRGT. (WING)
SURFACE JOINTS REPRESENTATIVE PANELS
②
②
③
BL 1.930 (76.00)
MLG WELL
40322
41236
40536
41036
41316
FS 4.064 (160.00)
FS 34.498 (1357.8)
FS 43.307 (1705.0)
FS 51.943 (2045.0)
FS 60.147 (2368.0)
FS 15.67 (617.00)
1
2
3
4
.5534 (21.789) C-C
.5512 (21.70) C-C
.5400 (21.25) C-C
.5469 (21.533) C-C
.5400 (21.25)
.593 (23.3
WING CARRY THRU
INCHES
0 100 200 400 600 800 1000 1200
0 5 10 15 20 25 30
METERS SCALE 1/100

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FOLDOUT FRAME 1

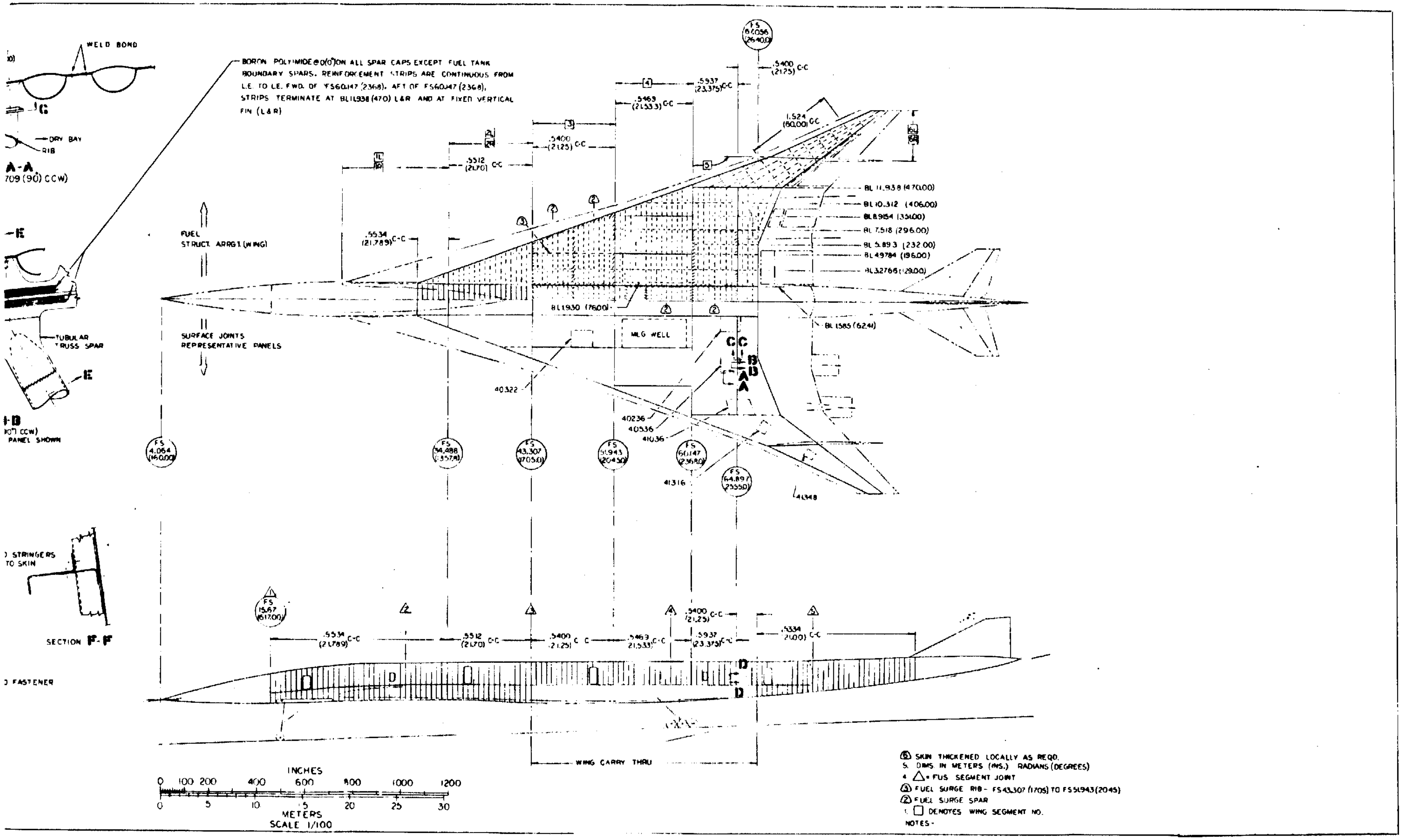
WELD BOND
DRY BAY
RIB
A-A
709 (90) CCW
TUBULAR TRUSS SPAR
PANEL SHOWN
STRINGERS TO SKIN
SECTION F-F
FASTENER
BORON POLYIMIDE @ 0(0°) ON ALL SPAR CAPS EXCEPT FUEL TANK BOUNDARY SPARS. REINFORCEMENT STRIPS ARE CONTINUOUS FROM L.E. TO L.E. FWD. OF FS60.147 (2368). AFT OF FS60.147 (2368), STRIPS TERMINATE AT BL11.938 (470) L&R AND AT FIXED VERTICAL FIN (L&R)
FUEL
STRUCT. ARRGT. (WING)
SURFACE JOINTS
REPRESENTATIVE PANELS
.5534 (21.789) C-C
.5512 (21.70) C-C
.5400 (21.25) C-C
.5463 (21.533) C-C
.5937 (23.375) C-C
.5400 (21.25) C-C
1.524 (60.00) C-C
.5334 (21.00) C-C
FS 4.064 (160.00)
FS 34.488 (1357.8)
FS 43.307 (1705.0)
FS 51.943 (2045.0)
FS 60.147 (2368.0)
FS 64.897 (2555.0)
FS 67.056 (2640.0)
FS 15.67 (617.00)
BL 11.938 (470.00)
BL 10.312 (406.00)
BL 8.954 (351.00)
BL 7.518 (296.00)
BL 5.893 (232.00)
BL 4.9784 (196.00)
BL 3.2766 (129.00)
BL 1.930 (76.00)
BL 1.585 (62.4)
MLG WELL
40322
40236
40536
41036
41316
41348
WING CARRY THRU
INCHES
0 100 200 400 600 800 1000 1200
0 5 10 15 20 25 30
METERS
SCALE 1/100
NOTES -
1. □ DENOTES WING SEGMENT NO.
② FUEL SURGE SPAR
③ FUEL SURGE RIB - FS43.307 (1705) TO FS51.943 (2045)
4. △ = FUS SEGMENT JOINT
5. DIMS IN METERS (INS.) RADIANS (DEGREES)
⑥ SKIN THICKENED LOCALLY AS REQD.

Figure 18-7. Composite Reinforced Wing Design - Fasteners

figures. The analytical studies reported in Section 12, Structural Concepts Analysis, resulted in a frame spacing of approximately 20 inches to yield minimum mass. Thus, for the chordwise stiffened wing design (including the composite reinforced spar caps design), the frames are located coincident with the spars. For the spanwise stiffened and monocoque wing designs, frames are located at various multiples of the spar spacing and are attached thereto; intermediate frames do not attach directly to the spars.

Weld bonding has been selected for the stringer/skin and frame/skin attachments. In this process, the basic spotweld attachment is made first and is followed by the infiltration of adhesive around the spotwelds. This enhances the fatigue properties of the joint in relation to a spotwelded joint and requires that individual fuselage panels be processed in curing ovens after being spotwelded together. Weld brazing is a backup for weld bonding if the weld bonding should prove unacceptable or impractical.

Since weld bonding is basically a spotwelding process, the panel size limits were established as a maximum of 15 feet in width. This width has been exploited in achieving a major advantage in that only one longitudinal skin panel splice, at the top centerline, is used in the cabin in the wing area. In the L-2000-7A design, spotwelding alone was used for skin/stringer and frame/stringer attachments in conjunction with skin panel splices at the 3 o'clock and 9 o'clock positions. At the top centerline and other panel splices, an external splice strip is used to provide a double shear joint with enhanced characteristics in fatigue. The top centerline splice in the cabin area is located at a plane of symmetry and minimizes the changes in skin thickness taking place at the splice since it means that thickness changes occur in the fore and aft direction only.

The fuselage segments are very close to 50 feet in length. This value is based on transportation considerations as well as the assumption of 50 feet long ovens for curing the adhesive used in the weld bonding and rivet bonding utilized extensively in fuselage fabrications. Furthermore, the fuselage is not broken at each wing segment. Instead, the fuselage is broken into approximately 50 foot long segments as discussed above. An important factor in the fuselage design is the fact that most, or all, of the fuselage is designed to permit subcontracting.

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Sandwich Shell Design — Welded

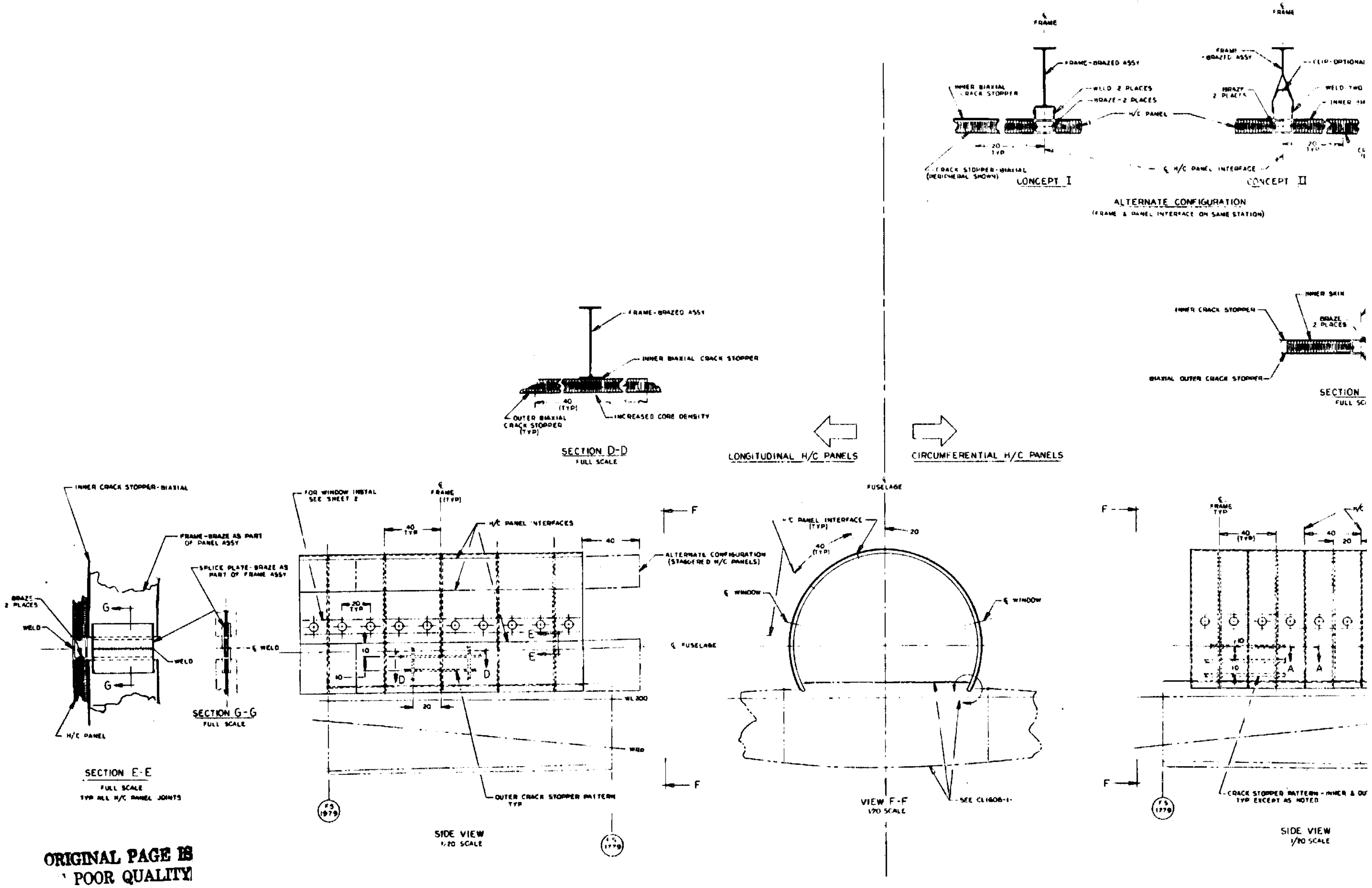
Various approaches for minimum mass fuselage designs were investigated. These studies were exploratory in nature but did identify the applicability of the concepts to a near-term start-of-design supersonic cruise aircraft.

Figure 18-8 presents one of the exploratory design studies made of sandwich fuselage structure. This particular study was based on the use of an aluminum brazed honeycomb sandwich shell ("Aeronca process") in conjunction with the welding joining method. It was recognized that the brazed sandwich results in a significant amount of parisitic weight of aluminum braze alloy, and thus increasing the difficulty of achieving a weight saving. However, the brazing process was assumed in this particular study because it offered the potential for structural integrity and reasonable adaptability to the welding joining method.

Two approaches are shown in Figure 18-8. The right side of the figure shows a design using circumferential panels which are continuous between floor/shell intersections. The design attempts to incorporate as many elements as possible within panel brazements to reduce assembly labor and joints. Panel brazements are EB welded together between frames and contain crack stoppers brazed to both skins. As shown, various frame designs and frame/shell joint designs were considered.

The left side of Figure 18-8 illustrates an approach using longitudinal panel brazements welded together with fore and aft EB welds. A panel width of 40 inches was assumed to minimize the contour deviation from a flat plane within each panel to reduce the problem with the braze alloy's tendency to run to the low point of the panel. Two variations of this design are shown. In one, circumferential panel joints are in a common fuselage station plane. In the other, circumferential panel joints are offset and not in a common fuselage station plane. This complicates the welding procedure but avoids a continuous welded joint in a single plane.

Although the studies showed the need for further development in several areas, the longitudinal panel design appears to be feasible and have some attractive features. Welded joints must be located carefully to avoid heat damage to the panels. The latter problem would be even more critical with an adhesive bonded/welded construction. An adhesive bonded circumferential panel design could, however, use



℄ FRAME
FRAME-BRAZED ASSY
INNER BIAXIAL CRACK STOPPER
WELD 2 PLACES
BRAZE-2 PLACES
H/C PANEL
20 TYP
CRACK STOPPER-BIAXIAL (PERIPHERAL SHOWN)
CONCEPT I
℄ FRAME
FRAME-BRAZED ASSY
CLIP-OPTIONAL
BRAZE 2 PLACES
WELD TWO
INNER
20 TYP
℄ H/C PANEL INTERFACE
CONCEPT II
ALTERNATE CONFIGURATION
(FRAME & PANEL INTERFACE ON SAME STATION)
INNER SKIN
INNER CRACK STOPPER
BRAZE 2 PLACES
BIAXIAL OUTER CRACK STOPPER
SECTION
FULL SC
FRAME-BRAZED ASSY
INNER BIAXIAL CRACK STOPPER
40 (TYP)
OUTER BIAXIAL CRACK STOPPER (TYP)
INCREASED CORE DENSITY
SECTION D-D
FULL SCALE
LONGITUDINAL H/C PANELS
CIRCUMFERENTIAL H/C PANELS
INNER CRACK STOPPER-BIAXIAL
FRAME-BRAZE AS PART OF PANEL ASSY
SPLICE PLATE-BRAZE AS PART OF FRAME ASSY
BRAZE 2 PLACES
WELD
G
G
WELD
℄ WELD
H/C PANEL
SECTION G-G
FULL SCALE
SECTION E-E
FULL SCALE
TYP ALL H/C PANEL JOINTS
FOR WINDOW INSTAL SEE SHEET 2
℄ FRAME (TYP)
40 TYP
H/C PANEL INTERFACES
40
20 TYP
ALTERNATE CONFIGURATION (STAGGERED H/C PANELS)
E
E
10
10
D
D
20
℄ FUSELAGE
WL 200
WRP
F
F
OUTER CRACK STOPPER PATTERN TYP
FS 1979
FS 1779
SIDE VIEW
1/20 SCALE
℄ FUSELAGE
H/C PANEL INTERFACE (TYP)
20
40 (TYP)
℄ WINDOW
℄ WINDOW
SEE CL1606-1
VIEW F-F
1/20 SCALE
F
F
℄ FRAME TYP
40 (TYP)
40
20
H/C
10
10
A
A
CRACK STOPPER PATTERN - INNER & OU
TYP EXCEPT AS NOTED
FS 1779
SIDE VIEW
1/20 SCALE

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FOLDOUT FRAME 1

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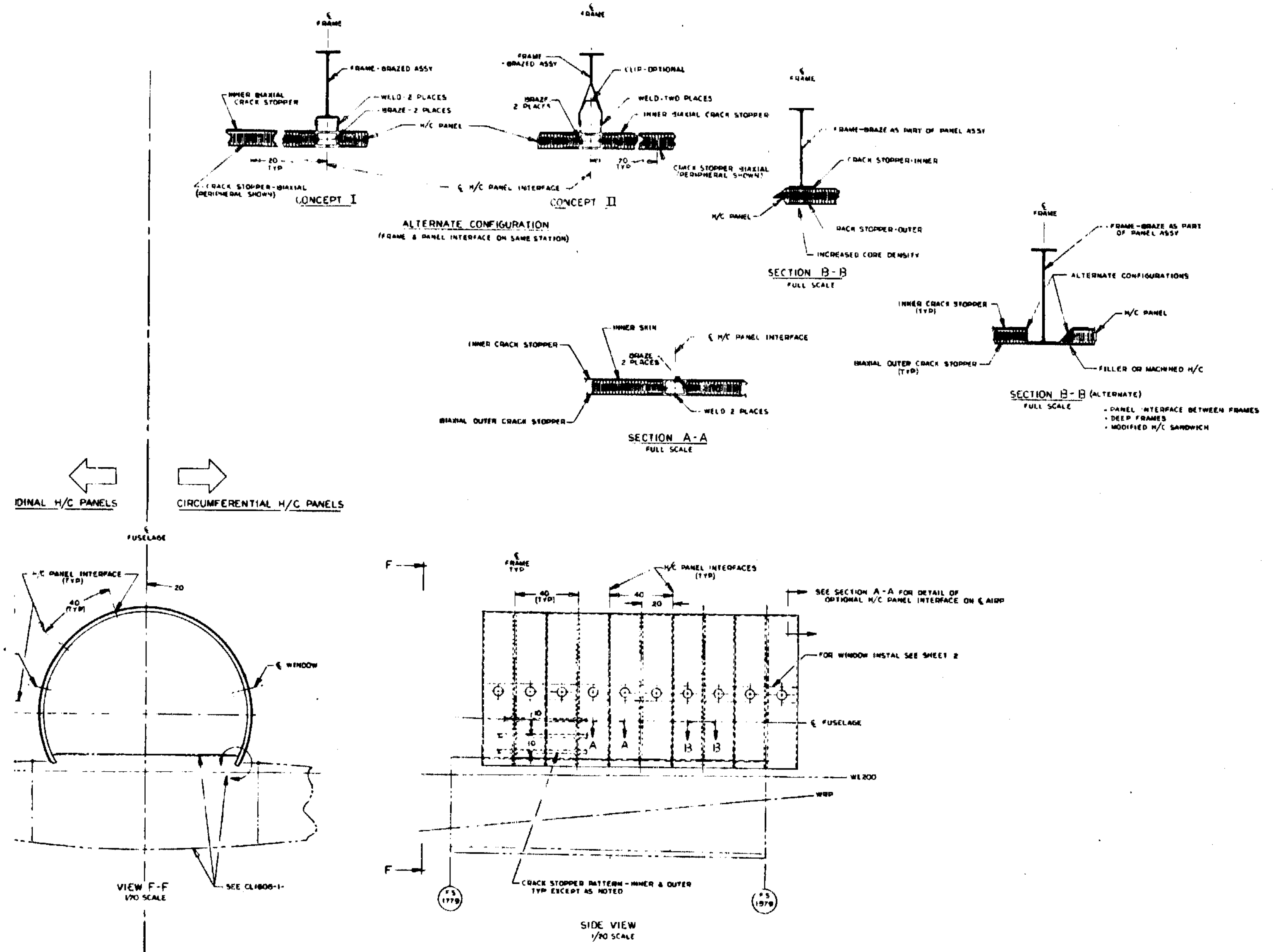
℄ FRAME
FRAME-BRAZED ASSY
INNER BIAXIAL CRACK STOPPER
WELD-2 PLACES
BRAZE-2 PLACES
H/C PANEL
20 TYP
CRACK STOPPER-BIAXIAL (PERIPHERAL SHOWN)
CONCEPT I
FRAME-BRAZED ASSY
CLIP-OPTIONAL
BRAZE 2 PLACES
WELD-TWO PLACES
INNER BIAXIAL CRACK STOPPER
20 TYP
CRACK STOPPER BIAXIAL (PERIPHERAL SHOWN)
℄ H/C PANEL INTERFACE
CONCEPT II
ALTERNATE CONFIGURATION
(FRAME & PANEL INTERFACE ON SAME STATION)
FRAME-BRAZE AS PART OF PANEL ASSY
CRACK STOPPER-INNER
H/C PANEL
CRACK STOPPER-OUTER
INCREASED CORE DENSITY
SECTION B-B
FULL SCALE
FRAME-BRAZE AS PART OF PANEL ASSY
ALTERNATE CONFIGURATIONS
INNER CRACK STOPPER (TYP)
H/C PANEL
BIAXIAL OUTER CRACK STOPPER (TYP)
FILLER OR MACHINED H/C
SECTION B-B (ALTERNATE)
FULL SCALE
• PANEL INTERFACE BETWEEN FRAMES
• DEEP FRAMES
• MODIFIED H/C SANDWICH
INNER SKIN
INNER CRACK STOPPER
℄ H/C PANEL INTERFACE
BRAZE 2 PLACES
WELD 2 PLACES
BIAXIAL OUTER CRACK STOPPER
SECTION A-A
FULL SCALE
DINAL H/C PANELS
CIRCUMFERENTIAL H/C PANELS
℄ FUSELAGE
H/C PANEL INTERFACE (TYP)
40 (TYP)
20
℄ WINDOW
SEE CL1606-1-
VIEW F-F
1/70 SCALE
F
℄ FRAME TYP
40 (TYP)
H/C PANEL INTERFACES (TYP)
40
20
10
A
B
SEE SECTION A-A FOR DETAIL OF OPTIONAL H/C PANEL INTERFACE ON ℄ AIRP
FOR WINDOW INSTAL SEE SHEET 2
℄ FUSELAGE
WL200
WRP
CRACK STOPPER PATTERN-INNER & OUTER TYP EXCEPT AS NOTED
FS 1779
FS 1979
SIDE VIEW
1/70 SCALE

4. MAX SKIN WIDTH ASSUMED TO BE 40 INCH
3. H/C PANEL & FRAME JOINED BY AL BRAZING
2. H/C PANELS JOINED BY WELDING
1. AL BRAZED TI[illegible] FRAMES & H/C PANELS

NOTE:

Figure 18-8. Sandwich Shell Design - Welded

considerably larger panels than those shown in Figure 18-8 for the brazed design. The preliminary studies of sandwich fuselage structure indicated considerable risk in conjunction with a limited weight saving potential relative to semi-monocoque design.

Design Problem Areas

A wide variety of structural designs were developed for evaluation and potential application in the wing and fuselage of the arrow-wing configuration supersonic cruise aircraft. Further design studies were conducted to evaluate the three wing structural arrangement and variations thereof. Candidate fuselage arrangements were also evaluated. These studies together with related design and manufacturing data, the manufacturing plan and the numerous drawings were developed to represent the major output of the design task.

One of the goals of this study is the identification of problem areas. Problem areas associated with design are briefly noted below for the various types of structure:

- Chordwise Stiffened

  a. With beaded outer skin, especially convex beads, thermal gradients and thermal cycling may cause disbonding at outer/inner skin weld bonded attachments. Loss of the bond would degrade fatigue and sealing properties. Therefore, verification tests are recommended.

  b. Convex beads in the outer skin, with thin gauges, cause concern relative to damage from hail, maintenance operations, FOD and other causes.

  c. Venting of inner/outer skin cavities at beads may present problems, particularly on the upper surface. One plan is to drill a 0.125 inch diameter hole in the outer skin at each end of a bead cavity. This might permit entry and accumulation of dirt and liquids over a long period and may require periodic cleaning of these cavities.

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d. Some concern is felt that the installation of fasteners-especially upset types-may cause disbonding even though a preliminary test indicates this may not be a problem.

e. Fretting of fastened joints is a concern.

f. The drag/fuel penalty associated with external beads in the outer skin has not been fully assessed.

g. When local thickening of the outer skin is used to avoid feather edges at countersinks in the outer skin, penalties in flushness/smoothness may result along with a significant cost increase.

h. Blind fasteners may be indicated in some areas as a result of a desire to minimize access doors.

i. The shape and size of submerged spar caps make it difficult to install surface/spar fasteners and constrain the type of fasteners.

j. Not all surfaces of all welds can be dressed and visually inspected. An example is a truss diagonal-to-end-fitting weld.

k. It is not feasible to stress relieve all weldments.

l. Fail-safe design of spar caps is difficult with this concept because of the monolithic nature and the large loads concentrated in spar caps. However, preliminary analyses indicate that the problem is soluble (without excessive mass penalty or complex measures) by exploiting load redistribution principles. Also, there may be a potential application for laminated structure for fail-safe metallic design.

- Spanwise Stiffened — See items d, e, g, h, j, k under the "Chordwise Stiffened" heading:

  a. As noted in (a) of the Chordwise Stiffened discussion above, thermal gradients and thermal cycling may cause disbonding at the outer/inner skin weld bonded attachments. If suitable tests show weld bonding is not acceptable, weld brazing is a possible backup approach.

- Aluminum Brazed Honeycomb Sandwich (Monocoque)
  a. This type of surface includes a significant parasitic weight of aluminum braze alloy. This also increases heat conduction through the sandwich. Development of "stop-off" or other fabrication techniques to minimize these problems is desirable.
  b. Achievement of reliable faying surface brazed joints may present problems, particularly if the width of the brazed joint is significant.
  c. Data are needed relative to the effectiveness of welded joints in crack-arresting applications.
  d. The prevention of excessive creep at brazed joints under long-term/high temperature conditions results in allowables much lower than the static strength.

DETAILED DESIGN — TASK II

Design Objectives

The detailed design studies were directed towards the final definition of the structural design for the wing and fuselage of the Mach 2.7 arrow-wing supersonic cruise aircraft. Other goals of this effort are the identification of:

- Problems not resolved during the study
- Components warranting further evaluation by fabrication and test.

Scope of Design Studies

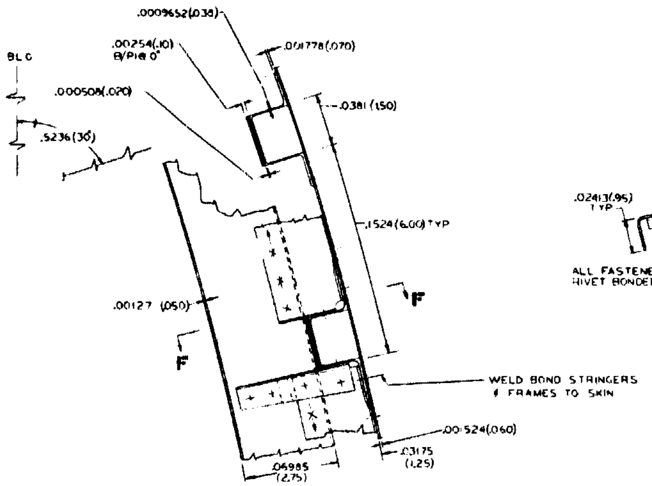
The Task I results reported in Section 17, Concept Evaluation and Selection identified the least mass and cost wing structure design to be the composite reinforced approach. The evaluation data also indicated the importance of minimum mass designs. Thus, a hybrid wing design was adopted for the detailed design studies effort. The hybrid design was developed by making full use of the minimum mass arrangement defined for the forward wing box, the aft wing box and the wing tip structure. The resulting minimum mass structural arrangement uses sandwich surfaces in the wing tip box and chordwise-stiffened surfaces with composite reinforced spar caps in the remainder of the wing box. The preliminary design drawings presented herein define the salient features listed below as applicable to both types of wing surface construction and the joint area between them:

- The overall structural arrangement for both the wing box and fuselage
- Surface and spar construction, as well as damage tolerance features, in the chordwise and monocoque and transition areas
- Fuselage structure design details for the skin/stringer/frame structural arrangement
- Composite reinforcement applications in the wing box and fuselage
- Manufacturing breakdown for the wing box and fuselage
- Fuel sealing provisions at tank wall spars and ribs and at intersections of composite reinforced spar caps with tank wall ribs

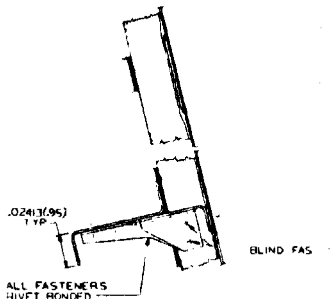
Wing Structure Design

The structural arrangement, manufacturing breakdown and the locations of representative panels and surface panel joints are presented in Figure 18-9 for the final design airplane. A hybrid structure consisting of the least mass chordwise stiffened - composite reinforced wing design and the monocoque wing design adopted for the final design are defined below:

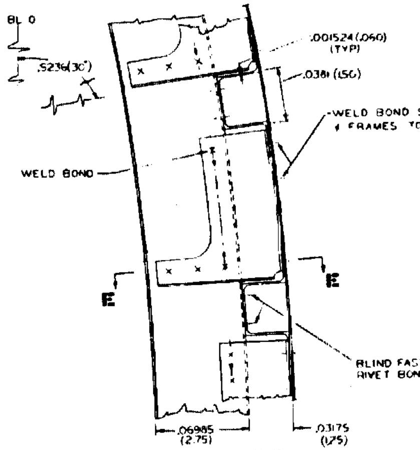
Wing Structure — Inboard of BL 470L/R — The design details for a specific chordwise-stiffened surface panel and substructure are shown at the right of Figure 18-10. With this beaded skin design, wing bending material is concentrated in the spar caps and the surfaces primarily transmit the chordwise and shear inplane loads. The uncoupling of bending and torsional material permits either the bending or torsional stiffness to be varied independently of the other. Also, this surface design alleviates thermal stresses and reduces heat transfer to the fuel, relative to a flat skin, since only a portion of the fuel is in direct contact with the wing external skin.


.0009652(.038)
.00254(.10) B/P1@0°
.001778(.070)
BL 0
.000508(.020)
.0381 (.150)
.5236(30°)
.1524(6.00) TYP
F
.00127 (.050)
F
WELD BOND STRINGERS & FRAMES TO SKIN
.001524(.060)
.03175 (1.25)
.06985 (2.75)

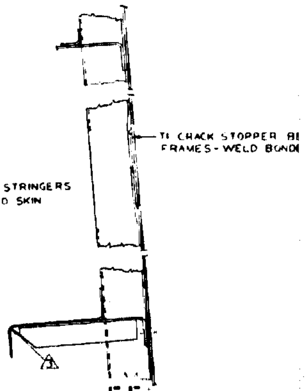
ALTERNATE VIEW D-D
COMPOSITE REINFORCED DESIGN


.02413(.95) TYP
BLIND FAS
ALL FASTENERS RIVET BONDED

SECTION F-F


BL 0
.001524(.060) (TYP)
.5236(30°)
.0381 (.150)
WELD BOND STRINGERS & FRAMES TO SKIN
WELD BOND
E
E
BLIND FASTENERS RIVET BONDED
.06985 (2.75)
.03175 (1.25)

VIEW D-D
METAL DESIGN


TI CRACK STOPPER BI
FRAMES-WELD BOND
A

SECTION E-E

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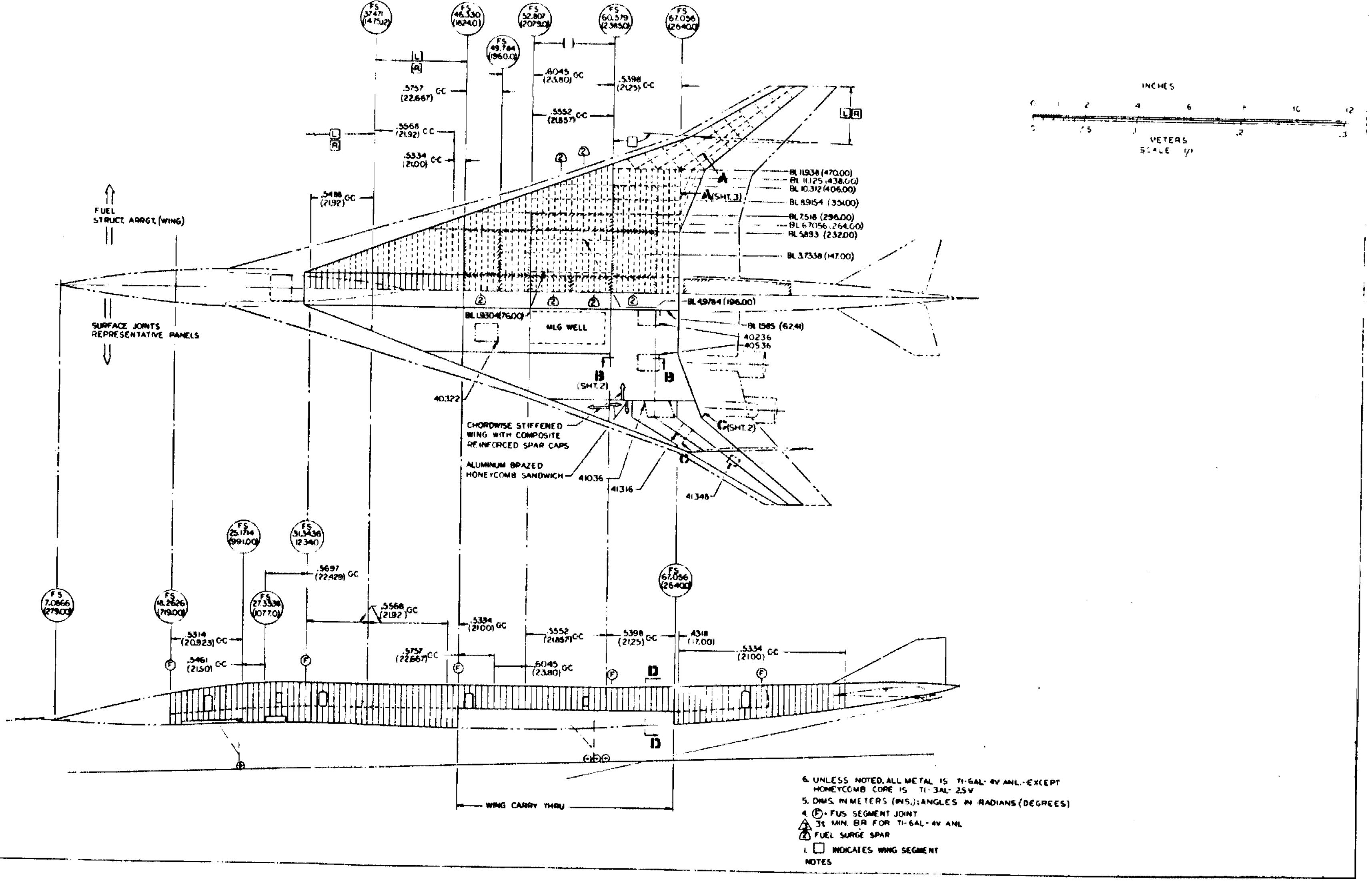
OF POOR QUALITY

FOLDOUT FRAME 1

DOUT FRAME 2

TENERS

TWEEN
D TO SKIN


FS 37.471 (1475.2)
FS 46.330 (1824.0)
FS 49.784 (1960.0)
FS 52.807 (2079.0)
FS 60.579 (2385.0)
FS 67.056 (2640.0)
L
R
.5757 (22.667) CC
.5568 (21.92) CC
.5334 (21.00) CC
.5488 (21.92) CC
.6045 (23.80) CC
.5552 (21.857) CC
.5398 (21.25) CC
INCHES
0 1 2 4 6 8 10 12
0 .05 .1 .2 .3
METERS
SCALE 1/1
FUEL
STRUCT ARRGT (WING)
SURFACE JOINTS
REPRESENTATIVE PANELS
A
A (SHT. 3)
BL 11.938 (470.00)
BL 11.125 (438.00)
BL 10.312 (406.00)
BL 8.9154 (351.00)
BL 7.518 (296.00)
BL 6.7056 (264.00)
BL 5.893 (232.00)
BL 3.7338 (147.00)
BL 4.9784 (196.00)
BL 1.9304 (76.00)
BL 1.585 (62.41)
MLG WELL
40236
40536
40322
B
B (SHT. 2)
C (SHT. 2)
C
CHORDWISE STIFFENED
WING WITH COMPOSITE
REINFORCED SPAR CAPS
ALUMINUM BRAZED
HONEYCOMB SANDWICH
41036
41316
41348
FS 7.0866 (279.00)
FS 18.2626 (719.00)
FS 25.1714 (991.00)
FS 27.3558 (1077.0)
FS 31.3436 (1234.0)
FS 67.056 (2640.0)
.5314 (20.923) CC
.5461 (21.50) CC
.5697 (22.429) CC
.5568 (21.92) CC
.5757 (22.667) CC
.5334 (21.00) CC
.5552 (21.857) CC
.6045 (23.80) CC
.5398 (21.25) CC
.4318 (17.00)
.5334 (21.00) CC
D
D
WING CARRY THRU

6. UNLESS NOTED, ALL METAL IS Ti-6AL-4V ANL. EXCEPT HONEYCOMB CORE IS Ti-3AL-2.5V
5. DIMS. IN METERS (INS.); ANGLES IN RADIANS (DEGREES)
4. Ⓕ - FUS SEGMENT JOINT
3. 3t MIN. BR FOR Ti-6AL-4V ANL
2. FUEL SURGE SPAR
1. ☐ INDICATES WING SEGMENT

NOTES

Figure 18-9. Structural Design Concepts - Surface Panels and Substructure

FOLDOUT FRAME 3

Weld bonding is the basic choice for joining the inner and outer skins of the surface assembly. Surface panel size has been held to 15 feet by 35 feet. The length limit is based on tooling considerations for hot vacuum forming of the skins while the width limit is based on the postulated size of spotwelding equipment.

Concern over potential damage from hail, FOD and maintenance operations has resulted in establishment of the following minimum gages:

- 0.020 in. for exterior skins on the wing lower surface
- 0.015 ins. for exterior skins on the wing upper surface
- 0.010 in. for inner skins of chordwise or biaxially stiffened surfaces

In locating wing spars in the chordwise-stiffened wing area, a minimum spacing of 21 inches has been maintained between constraints such as fuel tank boundaries. This spacing, at or slightly above the optimum determined in Task I, is based on providing a minimum of 19 inches clear passage between spars. This passage width, with sufficient depth, allows internal movement of personnel for installing surface attachment fasteners and performing inspection and maintenance operations. This spacing approach has been taken to avoid blind fasteners and minimize access doors.

The relatively thin external skin in the weld bonded surfaces, in conjunction with flush fasteners, result in feather edges at countersinks in some areas. Where this is unacceptable, thickened pads are incorporated in the skin. Section I-I of Figure 18-10 depicts an example of such a spar/surface attachment. As shown, diffusion bonding is specified as an alternate method for joining the inner and outer skins. Low pressure diffusion bonding is a very attractive process which, when adequately developed, would avoid or minimize the need for thickened pads in the outer skin and would thereby permit reductions in mass and surface roughness. If attachment clips are included in such a diffusion bonded surface assembly, the assembly could be fastened to a welded truss spar as sketched in Figure 18-11. By reducing fasteners through the surface, this approach would facilitate fuel sealing also.

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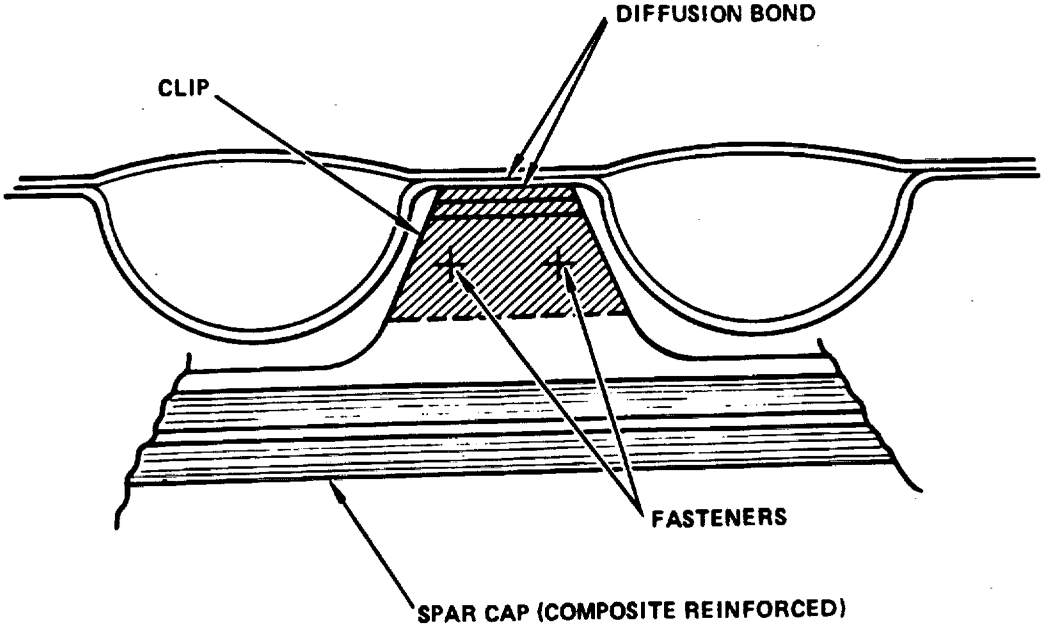
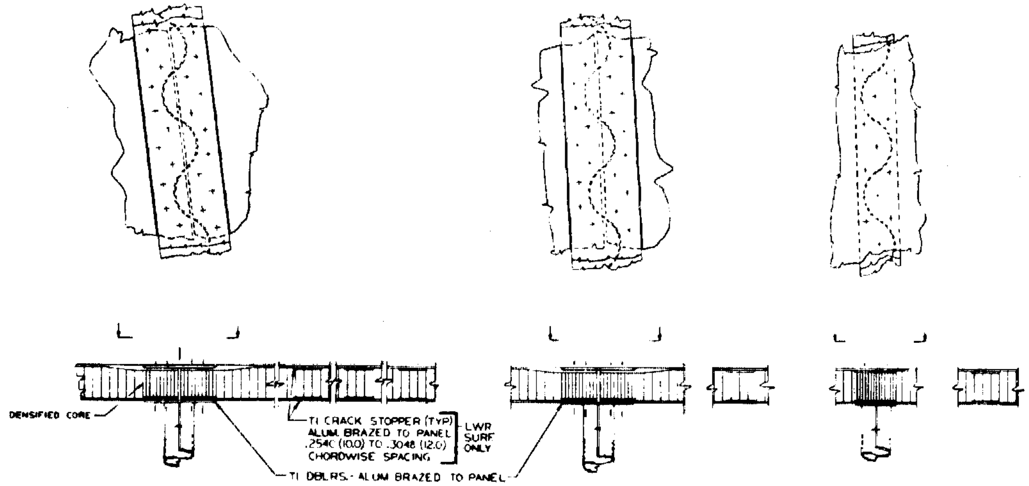


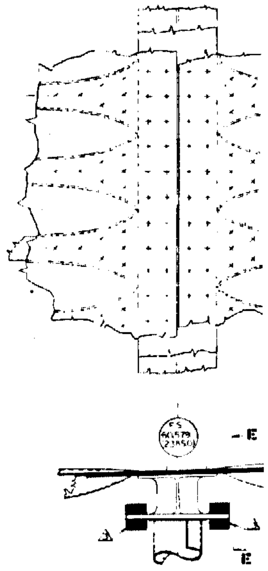
Figure 18-11. Attachment of Truss Spar to Low Pressure Diffusion Bonded Surface

Wing rib spacing is a nominal 60 inches but is modified as required to suit design constraints. In the chordwise-stiffened and transition areas, welded truss spars are used except where a spar serves as a fuel tank wall. At such locations, spars have welded circular arc webs with stiffened "I" caps. To facilitate fuel sealing, no surface beads extend across tank boundaries. Inner wing spars in the aft wing box are fabricated initially as continuous subassemblies between BL 470L and R. At the latter locations, as shown in Figure 18-12 spars are subsequently welded together to form continuous spars from one vertical fin to the opposite. An 85-foot long vacuum chamber has been postulated for electron beam welding of spar subassemblies. Whenever possible, all welds are designed for automation as well as stress relieving, dressing on all surfaces and visual inspection after welding.

Effective use is made of composite reinforcement with the major application involving unidirectional reinforcement of spar caps. At all welded truss spars, the composite reinforcement is continuous between BL 470L and R. Figure 18-12 details the outboard termination of the spar cap reinforcements. The continuity of these reinforcements is very desirable even though they penetrate fuel tank wall ribs. The design developed for fuel sealing at intersections of composite reinforced welded truss spars with


DENSIFIED CORE
TI CRACK STOPPER (TYP)
ALUM. BRAZED TO PANEL
.2540 (10.0) TO .3048 (12.0)
CHORDWISE SPACING
LWR SURF ONLY
TI DBLRS.- ALUM BRAZED TO PANEL

SECTION C-C (FROM SHT. I)
ROTATED .9250 (53°)


F.S.
601579
(23850)
E
E

FOLDOUT FRAME 1

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ORIGINAL PAGE IS
OF POOR QUALITY

FOLDOUT FRAME 2

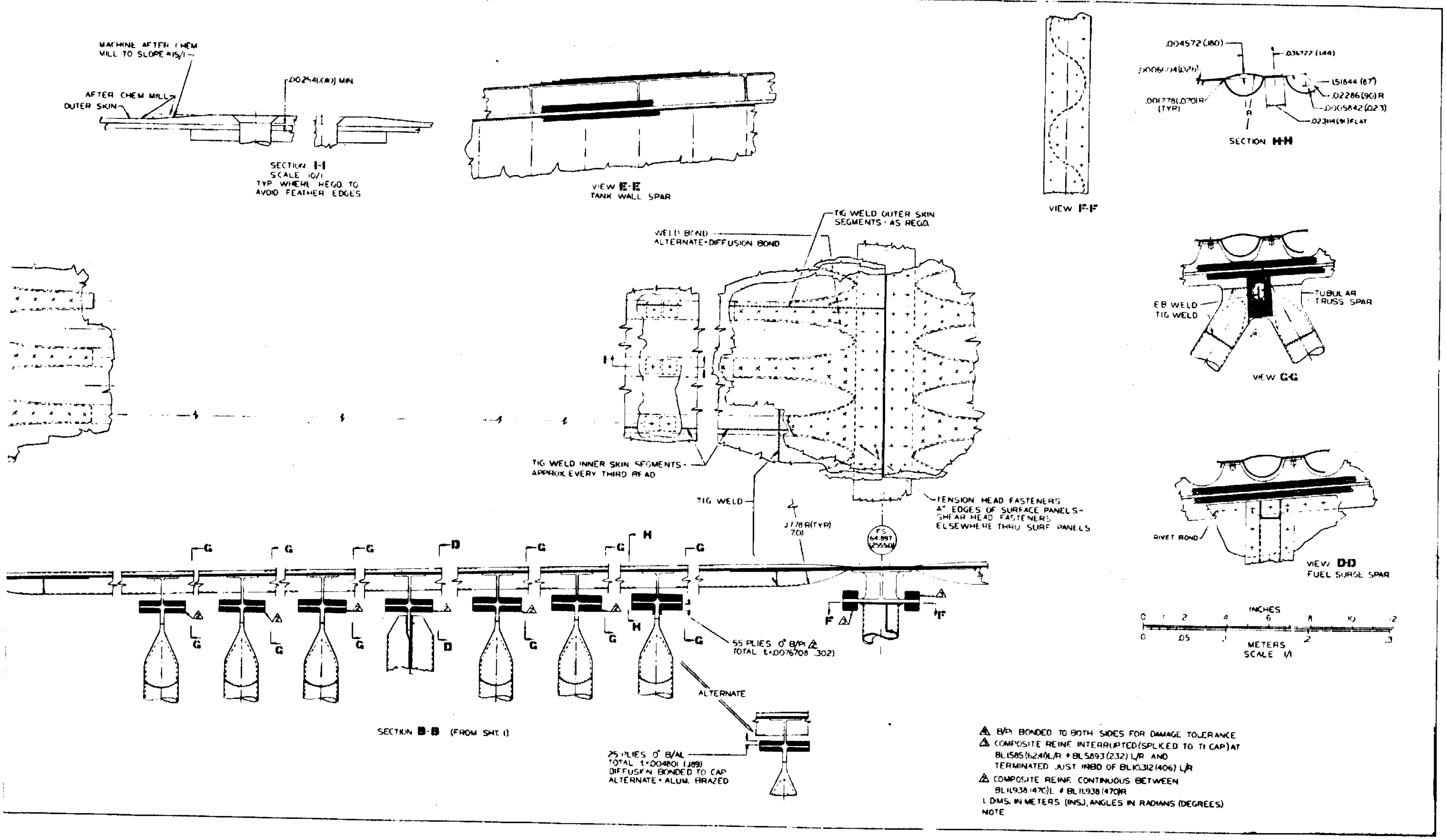
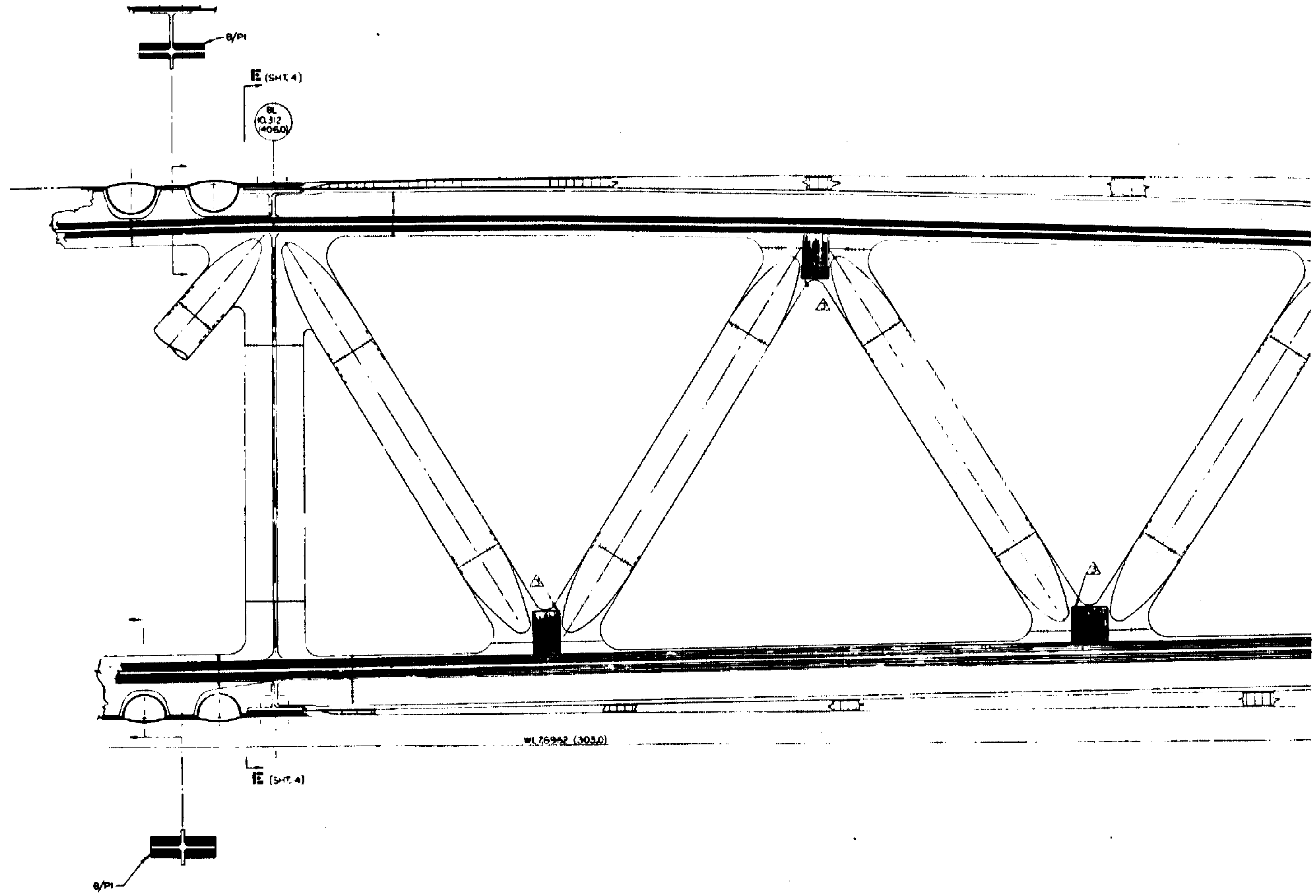

MACHINE AFTER CHEM MILL TO SLOPE ≈15/1
AFTER CHEM MILL
OUTER SKIN
.00254(.100) MIN
SECTION I-I
SCALE 10/1
TYP WHERE REQD TO AVOID FEATHER EDGES
VIEW E-E
TANK WALL SPAR
VIEW F-F
.004572(.180)
.036722(.144)
.00066/04(.026)
.001778(.070)R (TYP)
.051844(.87)
.02286(.90)R
.0005842(.023)
.02314(.91)FLAT
R
SECTION H-H
TIG WELD OUTER SKIN SEGMENTS - AS REQD
WELD BOND
ALTERNATE - DIFFUSION BOND
EB WELD
TIG WELD
TUBULAR TRUSS SPAR
VIEW G-G
TIG WELD INNER SKIN SEGMENTS - APPROX EVERY THIRD READ
TIG WELD
.1778R(TYP) (7.0)
FS 64.897 (2555.0)
TENSION HEAD FASTENERS AT EDGES OF SURFACE PANELS - SHEAR HEAD FASTENERS ELSEWHERE THRU SURF PANELS
RIVET BOND
VIEW D-D
FUEL SURGE SPAR
INCHES
0 1 2 4 6 8 10 12
0 .05 .1 .2 .3
METERS
SCALE 1/1
G
H
D
F
55 PLIES 0° B/PI 2
TOTAL t=.0076708 (.302)
ALTERNATE
SECTION B-B (FROM SHT. 1)
25 PLIES 0° B/AL
TOTAL t=.004801 (.189)
DIFFUSION BONDED TO CAP
ALTERNATE - ALUM. BRAZED
4 B/PI BONDED TO BOTH SIDES FOR DAMAGE TOLERANCE
3 COMPOSITE REINF INTERRUPTED (SPLICED TO TI CAP) AT BL1585(62.40)L/R & BL5893(232) L/R AND TERMINATED JUST INBD OF BL10.312(406) L/R
2 COMPOSITE REINF. CONTINUOUS BETWEEN BL11.938(470)L & BL11.938(470)R
1 DIMS. IN METERS (INS), ANGLES IN RADIANS (DEGREES)
NOTE

Figure 18-10. Structural Design Concepts-
Truss Spar and Surface Panel
at Transition Structure


B/P1
E (SHT. 4)
BL
10.312
(406.0)
3
3
3
WL 76.962 (303.0)
E (SHT. 4)
B/P1

FOLDOUT FRAME 1

FOLDOUT FRAME 2

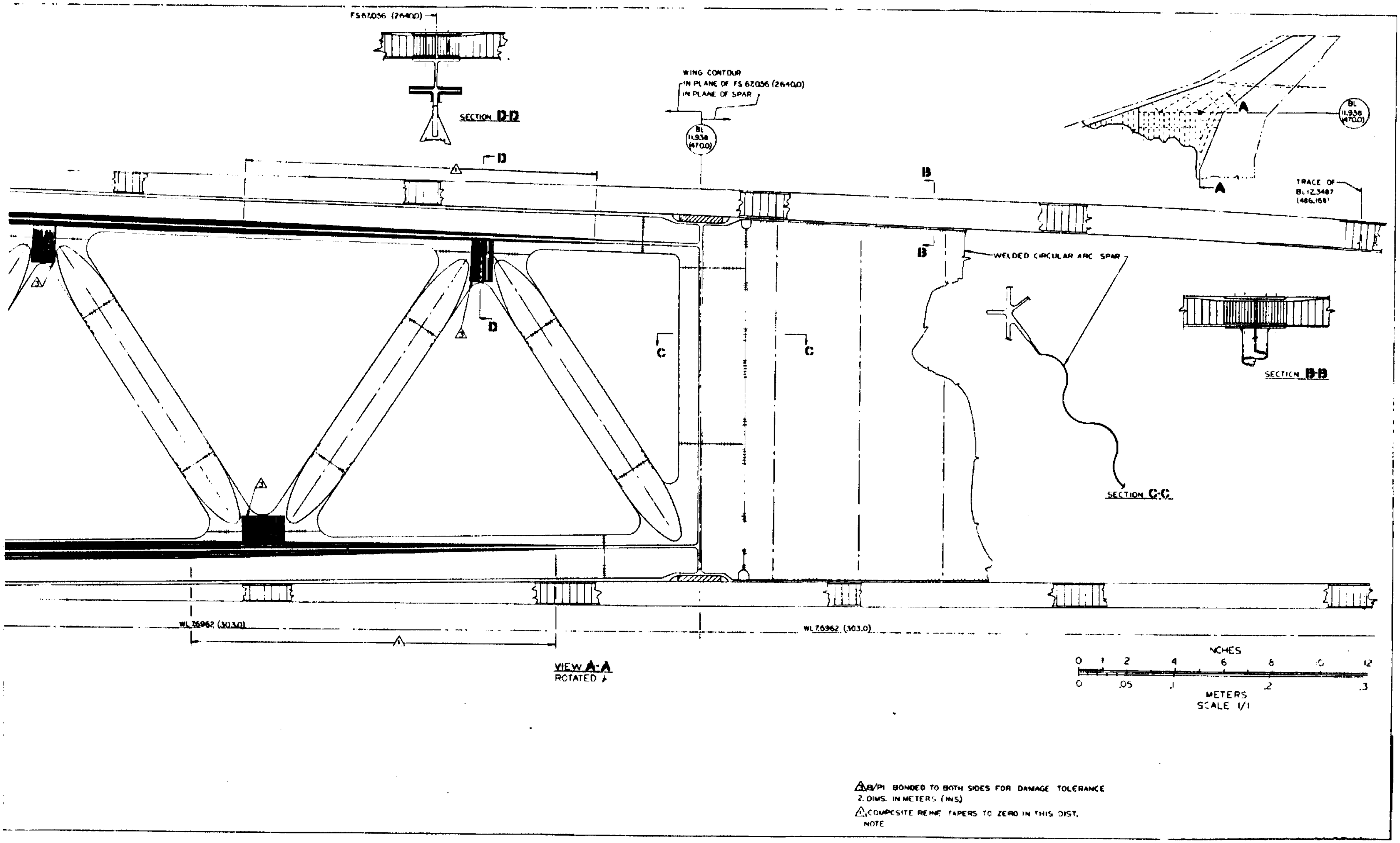

FS 67.056 (2640.0)
SECTION D-D
WING CONTOUR
IN PLANE OF FS 67.056 (2640.0)
IN PLANE OF SPAR
BL 11.938 (470.0)
D
D
1
3
3
B
B
A
A
BL 11.938 (470.0)
TRACE OF BL 12.3487 (486.168)
WELDED CIRCULAR ARC SPAR
SECTION B-B
C
C
SECTION C-C
WL 7.6962 (303.0)
WL 7.6962 (303.0)
1
VIEW A-A
ROTATED
INCHES
0 1 2 4 6 8 10 12
0 .05 .1 .2 .3
METERS
SCALE 1/1
3 B/PI BONDED TO BOTH SIDES FOR DAMAGE TOLERANCE
2. DIMS. IN METERS (INS)
1 COMPOSITE REINF TAPERS TO ZERO IN THIS DIST.
NOTE

Figure 18-12. Structural Design Concepts Tank Sealing Concept and Rib Details

tank wall ribs is shown at the left of Figure 18-13. Machined aluminum fittings, which are not attached through the composite, fit tightly around the reinforced spar caps to provide both fuel sealing and spar cap support. The right side of Figure 18-13 shows the intersection of a composite reinforced truss spar with BL 406 tank wall rib aft of the area where fuel sealing is required. At tank wall spars, as shown in Figure 18-10, the composite reinforcement consists of separate reinforcement strips between adjacent tank wall ribs. This interruption is used so that a special metal fitting, made an integral part of the spar cap, can be designed to provide an efficient structural joint between tank wall ribs and tank wall spars while also permitting a clean tank corner amenable to fuel sealing.

Boron/polyimide (B/PI) is specified as the basic composite material for spar cap reinforcement because of its lower cost and structural efficiency. The multiple element form of the B/PI spar cap reinforcement results in damage tolerance capability. Boron/aluminum (B/Al) diffusion bonded to spar caps is an alternate approach and one which has been used by Ameron. An additional alternate is the use of Borsic/aluminum reinforcements', which are aluminum brazed to the caps. In another extensive application, composite reinforcement is bonded to both sides of the welded diagonal-to-cap joints in all welded truss spars to serve as a crack stopper. This is depicted in View G-G of Figure 18-10 and Figure 18-12.

A tentative plan for venting the closed cavities at inner/outer skin beads is the use of a small diameter hole in the outer skin at each end of a bead cavity. For the upper surface in particular, this requires further review since this might permit entry and accumulation of dirt and liquids over a period of time.

Problem areas identified for the wing structural design are discussed in a later section.

Wing Structure — Outboard of BL 470L/R — Design details for the monocoque surfaces and the substructure in the wing tip box are defined at the left of Figure 18-9. The sandwich surfaces are brazed together using 3003 aluminum alloy as the brazing material (the "Aeronca" process). Outboard of BL 470, welded circular-arc spars and ribs are used since the minimum or zero need for web penetrations allows the realization of their inherent light weight and simplicity. Composite reinforcement is not used in the brazed surfaces or the welded circular arc spars and ribs.

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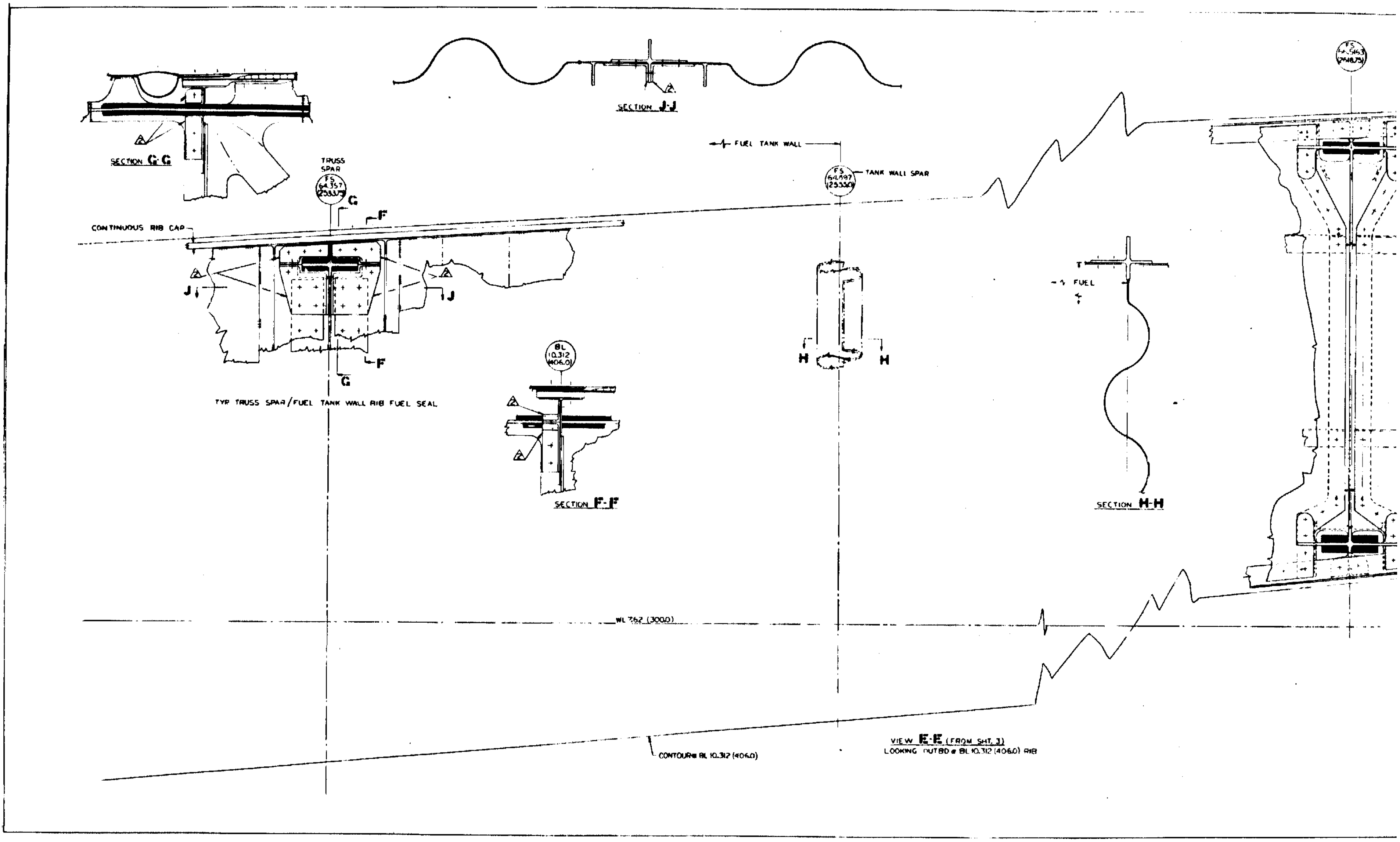
SECTION G-G
SECTION J-J
TRUSS SPAR
FS 64.357 (253.375)
G
F
CONTINUOUS RIB CAP
J
TYP TRUSS SPAR/FUEL TANK WALL RIB FUEL SEAL
BL 10.312 (406.0)
SECTION F-F
FUEL TANK WALL
FS 64.497 (253.50)
TANK WALL SPAR
H
T
℄ FUEL
SECTION H-H
FS 64.5463 (254.875)
WL 7.62 (300.0)
CONTOUR @ BL 10.312 (406.0)
VIEW E-E (FROM SHT. 3)
LOOKING OUTBD @ BL 10.312 (406.0) RIB

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FOLDOUT FRAME /

FOLDOUT FRA

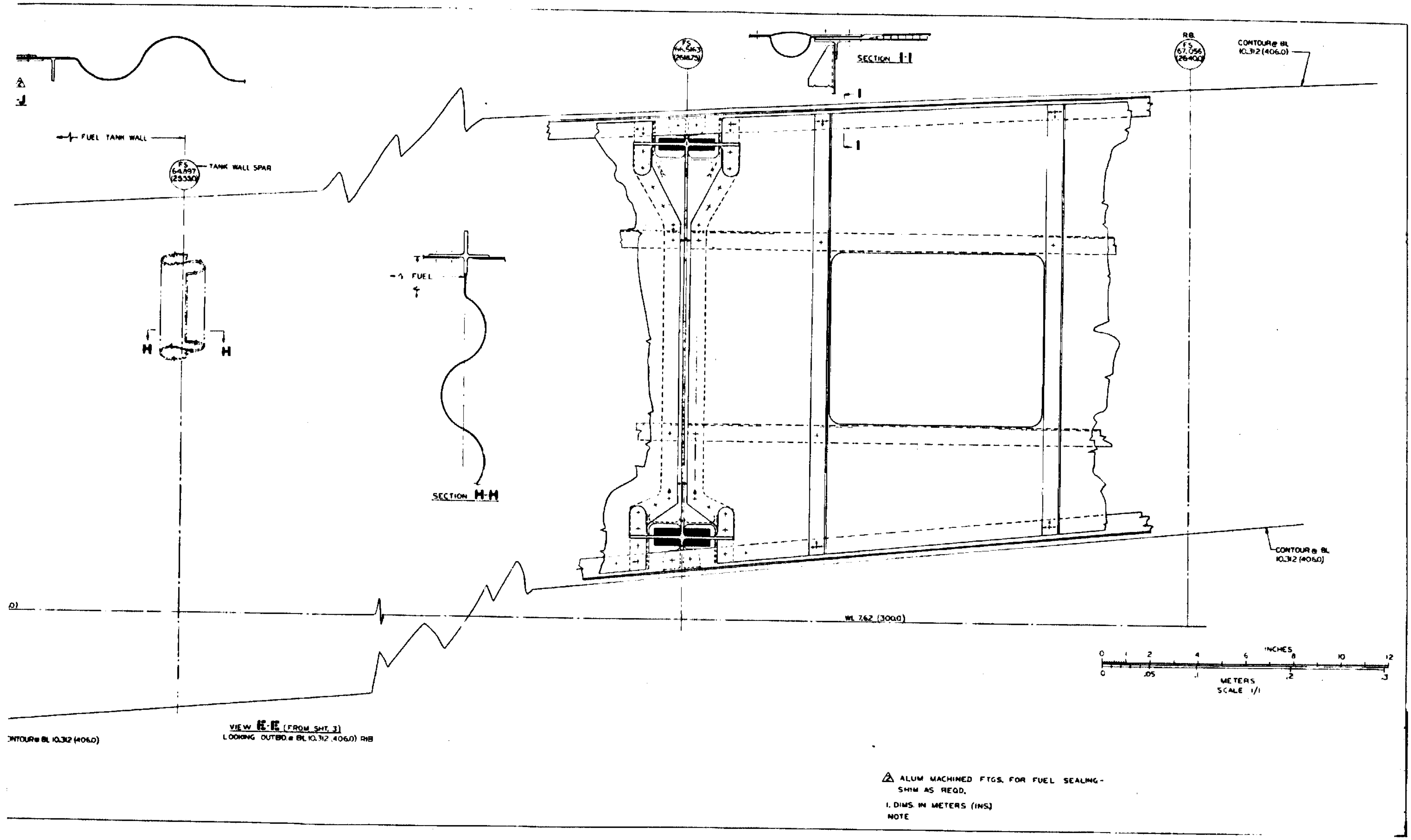
FS 66.5163 (2618.75)
SECTION I-I
I
I
R.B.
FS 67.056 (2640.0)
CONTOUR @ BL 10.312 (406.0)
FUEL TANK WALL
FS 64.897 (2555.0)
TANK WALL SPAR
H
H
FUEL
SECTION H-H
CONTOUR @ BL 10.312 (406.0)
WL 7.62 (300.0)
INCHES
0 2 4 6 8 10 12
0 .05 .1 .2 .3
METERS
SCALE 1/1
VIEW E-E (FROM SHT. 3)
LOOKING OUTBD. @ BL 10.312 (406.0) RIB
NTOUR @ BL 10.312 (406.0)
ALUM MACHINED FTGS. FOR FUEL SEALING-SHIM AS REQD.
1. DIMS. IN METERS (INS.)
NOTE

Figure 18-13. Structural Design Concept - M2.7 Arrow-Wing Supersonic Transport Configuration (Near-Term)

FOLDOUT FRAME 2

FOLDOUT FRAME 3

A size limit of 68 inches by 40 feet for brazed surfaces was postulated as a guide after consultation with Aeronca. The actual panel configurations defined in Figure 18-9 approximate this size and are based on the design philosophy that all or some panels of the upper surface are attached with screws and are removable for inspection and maintenance purposes.

The flexibility of the aluminum braze process is exploited by incorporating crack stoppers and panel edge doublers in the surface panel brazements. Also, the capability of tapering the panel thickness is utilized in the joint between the chordwise and monocoque surface areas. In the joint area, as shown in Figure 18-12, the outboard sandwich surfaces are extended inboard across BL 470 so that spanwise components of the outboard surface loads due to wing bending loads are transferred directly to the chordwise stiffened structure at the BL 406 rib.

Spanwise hat section stiffeners are aluminum brazed to the tapered wing surface panels in the joint area to provide the needed surface stiffening. Surface panel splice plates are sufficiently thick to avoid feather edges at countersunk fasteners.

Fuselage Structure Design

The fuselage structural arrangement and design details of a specific fuselage area for both all titanium and composite reinforced metal structure are shown in Figure 18-9. As shown, both designs include machined extrusion stringers, crack stoppers between frames and floating zee frames with shear clips. Closed hat section extruded stringers which provide structural efficiency, are machined to provide for crack stoppers and to vary stringer thickness. Extruded stringers also are well suited to effective installation of composite reinforcement. The floating zee frames with shear clips are considered preferable, from a fatigue standpoint, to full depth frames having notches for stringers. Also, zee frames avoid the offset shear center associated with channel section frames.

Weld bonding is used for attaching frames, stringers and crack stoppers to the skin because of economy, minimum mass, good fatigue characteristics and the avoidance of sealing problems. Staisfactory weld bonding of three thicknesses, as encountered at some locations, may require development. Weld brazing is a possible backup to weld bonding. Where fasteners are used at shear clips and frame/stringer

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attachments, fastener bonding is utilized in lieu of fasteners alone to obtain enhanced fatigue properties. The size of fuselage skin panel assemblies has been limited to 15 ft by 50 ft; the former is based on the postulated size of spot-welding equipment, the latter on the postulated length of the adhesive curing ovens.

Longitudinal skin panel splices are located only at the top and bottom centerlines of the fuselage and at the floor/shell intersections fore and aft of the wing carry-through area. These longitudinal splices utilize external and internal splice plates in conjunction with fastener bonding to achieve a double shear splice having damage tolerance capabilities and good fatigue properties. Suitable combinations of fastener size and external splice plate thickness are utilized to avoid feather edges at countersinks for flush fasteners. At circumferential panel splices, and other locations as required, feather edges are avoided by incorporating thickened pads in the external skin in a manner similar to that for wing skins. Chemical milling is used to vary fuselage skin thickness in accordance with load requirements.

Potential Problems Not Resolved During Study

Design and Manufacture

Preliminary design studies of a wide variety of structural design concepts were made to assess the relative merits of various concepts and materials suitable for an advanced supersonic cruise aircraft (Mach 2.7). Construction details were provided for wing and fuselage primary structure to enable mass and cost estimates to be made. In the process of preparation of the preliminary engineering drawings potential problems were uncovered and resolved (on the drawing) by applying good engineering judgement, related subscale experimental work and decisions made in consultation with Structures, Advanced Design, Materials and Producibility specialists.

The critical problems in the fabrication technology were joining, forming, sealing and related equipment requirements. These disciplines were further affected by the materials constraints. Some industry research and development programs have been initiated in these technologies, however, specific application to a supersonic cruise aircraft (Mach 2.7) with an arrow-wing design configuration presents many new problems to be resolved. The size effect of hardware components, for example, presents some of the critical fabrication problems.

A summary of the potential problem areas for design and manufacture of wing and fuselage primary structure are identified in Tables 18-3 through 18-7. These problem areas require further detailed study to develop the fabrication processes, tooling methods, and definition of the facilities to verify the manufacturing capability to produce hardware components that meet the engineering requirements.

TABLE 18-3. POTENTIAL PROBLEMS NOT RESOLVED - WING SURFACE PANEL CONCEPT - CONVEX BEADED

Wing Surface Panel Concept - Convex Beaded

- Welding of thin gage sheet to fabricate large panels (15 ft. x 35 ft.)
- Vacuum forming skin panels to final compound contour
- Continuous progressive rolling of inner skins
- Assembly of skin panels
    - Weld bonding
    - Weld brazing
    - Isothermal brazing
    - Diffusion bonding (low pressure)
- Fastening minimum gage skin panels to substructure
    - Disbonding at fasteners through weld bonded surfaces
    - Rivet types and installation clearance requirements
- Disbonding resulting from thermal gradients and cycling
- Venting of beaded skins
- Fail-safe characteristics and fatigue quality

TABLE 18-4. POTENTIAL PROBLEMS NOT RESOLVED DURING STUDY - WING SPAR CONCEPT - COMPOSITE REINFORCED TITANIUM ALLOY CAPS

Wing Spar Concept - Composite Reinforced Titanium Alloy Caps

- Boron/polyimide reinforcement
  - Autoclave and heat expanding rubber oven curing methods
  - Tooling methods
  - Process procedures
  - NDT methods
- Borsic/aluminum reinforcement
  - Aluminum brazing methods
- Boron/aluminum reinforcement
  - Diffusion bonding methods
- Boron/aluminum truss tubes
  - Diffusion bonding methods
  - Brazing methods
- Fail-safe characteristics

TABLE 18-5. POTENTIAL PROBLEMS NOT RESOLVED DURING STUDY - WING SURFACE PANEL CONCEPT - HONEYCOMB SANDWICH

Wing Surface Panel Concept - Honeycomb Sandwich

- Aluminum brazing large panels to compound contour
- Brazing doublers and fail-safe straps
- Core stop-off material and processes to prevent vertical cell wall braze flow during brazing cycle
- Braze panel structural repair methods
  - Potting compounds (organic plus ceramic materials)
  - Mechanical fastened patch plates
  - Tooling/equipment requirements
  - NDT techniques
- Fail-safe characteristics

TABLE 18-6. POTENTIAL PROBLEMS NOT RESOLVED DURING STUDY - FUSELAGE SHELL CONCEPT - SKIN AND STRINGER

Fuselage Shell Concept - Skin-Stringer

- Weld bonded panel assemblies (15 ft. x 50 ft.)
    - Skin and stringers
    - Frames
    - Fail-safe straps
    - Shear clips
- Rivet bond panel section splice
    - Longitudinal (50 ft. length)
    - Circumferential (204 in. length)
    - Equipment/processing techniques required
- Precision extruded titanium alloy sections
    - Hat section stringers
    - Variable wall thicknesses

TABLE 18-7. POTENTIAL PROBLEMS NOT RESOLVED DURING STUDY - TANK SEALING CONCEPT

Tank Sealing Concept

- Requirements
    - Fuel tank thermal environment
    - Application techniques
    - Curing methods
- Faying surface
    - Interstices between structural members
    - Corner gaps
    - Overlapping surfaces
    - Coating fastener, welds and pin hole openings
- Intersection of composite reinforced spar caps/tank rib caps
    - Sealing techniques
    - Metal fittings plus sealing material

Components for Further Evaluation and Test

The development of structural concepts for supersonic cruise aircraft has been the subject of this reported analytical investigation.

This investigation has produced a number of promising concepts having good potential application for design of a supersonic airplane structure. However, experimental research is necessary to assess the validity of the structural concepts and to determine the combination of concepts best suited for design and fabrication of flight hardware for Mach 2.7 arrow-wing configuration transport.

The economic viability of a supersonic cruise aircraft is dependent upon achievement of a low structural mass fraction through lightweight and reliable structural design concepts. State-of-the-art structures and materials are inadequate to provide a viable commercial supersonic transport with a service life of 50,000 hours.

The principal objective of this planned research and development program was to assess the relative merits of various structural concepts and materials for a prescribed arrow-wing aerodynamic configuration; to determine the structural weight estimates based on in-depth structural design studies. The concepts were evaluated through design studies making use of the best available materials technology, design tools, design criteria and simplified cost benefit studies. The best concepts which merit experimental evaluation are identified.

As an integral part of the research and development program it is essential that an experimental program be initiated to verify the manufacturing capability to produce hardware components that meet the engineering requirements as shown in Table 18-8. Moreover, experimental data verifying fabrication processes (Table 18-9) from the element and component tests (Table 18-10) must be compared with predicted values of strength, deflections, temperature distributions, fatigue quality, fail-safe characteristics, and life to determine the degree to which concept performance can be predicted in a realistic structural application. The results of these tests should then be used to refine the methods of analyses and concept designs.

It is recommended that future studies be directed towards the design, fabrication, and testing (with appropriate attachment and restraint conditions) under simulated flight environment of larger structural assembly incorporating the refined concept designs. The outer wing and nacells of the NASA YF-12 aircarft could be effectively used as the baseline structural envelope for design and fabrication of such an assembly. After sufficient ground testing, actual flight experience at the high Mach ranges can be obtained.

TABLE 18-8. TEST OBJECTIVES

Conduct Sufficient Tests to Experimentally Validate the Structural Design Concepts:

- Manufacturing processes
- Strength and damage tolerance capability of the components for the load/temperature environment
- Correlate test results with the theoretical analysis
- Assess the impact upon the results of this design concept study

TABLE 18-9. FABRICATION PROCESS TESTS

Types of Test Recommended

- Fabrication processes
    - Tooling approach
    - Fabrication and assembly schedule
    - Joining methods
    - Chemical milling/shot peening
    - Metallurgical examination
    - Inspection/quality assurance methods

TABLE 18-10. STRUCTURAL ELEMENT AND COMPONENT TESTS

Types of Test Recommended

- Structural element and component
  - Room temperature
  - Elevated temperature
  - Cyclic
- Development tests
  - Crippling
  - Buckling
  - Column
  - End closeout
  - Inplane shear
  - Thermal cycling
  - Mechanical joints - fatigue quality, fretting
  - Chem mill/shot peen - fatigue properties
  - Crack growth/propagation
  - Crack stopper concept
  - Fail-safe and fatigue characteristics
  - Welded joints - fatigue and fracture properties

REFERENCES

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2. Elrod, S.D. and Lovel, D.T: Development of Aluminum Brazed Titanium Honeycomb Sandwich Structure, Boeing Aircraft Report D6-60203, July 1972 (Report No. FAA-SS-72-03).

SECTION 19

PROPULSION-AIRFRAME INTEGRATION

BY

S.J.SMYTH, D.M.URIE, T.A.SEDGWICK, I.F.SAKATA

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SECTION 19

PROPULSION - AIRFRAME INTEGRATION

INTRODUCTION

The study conducted to provide the proper integration of the engine and the airframe for meaningful completion of the detailed engineering design and analysis of Task II is reported in this section.

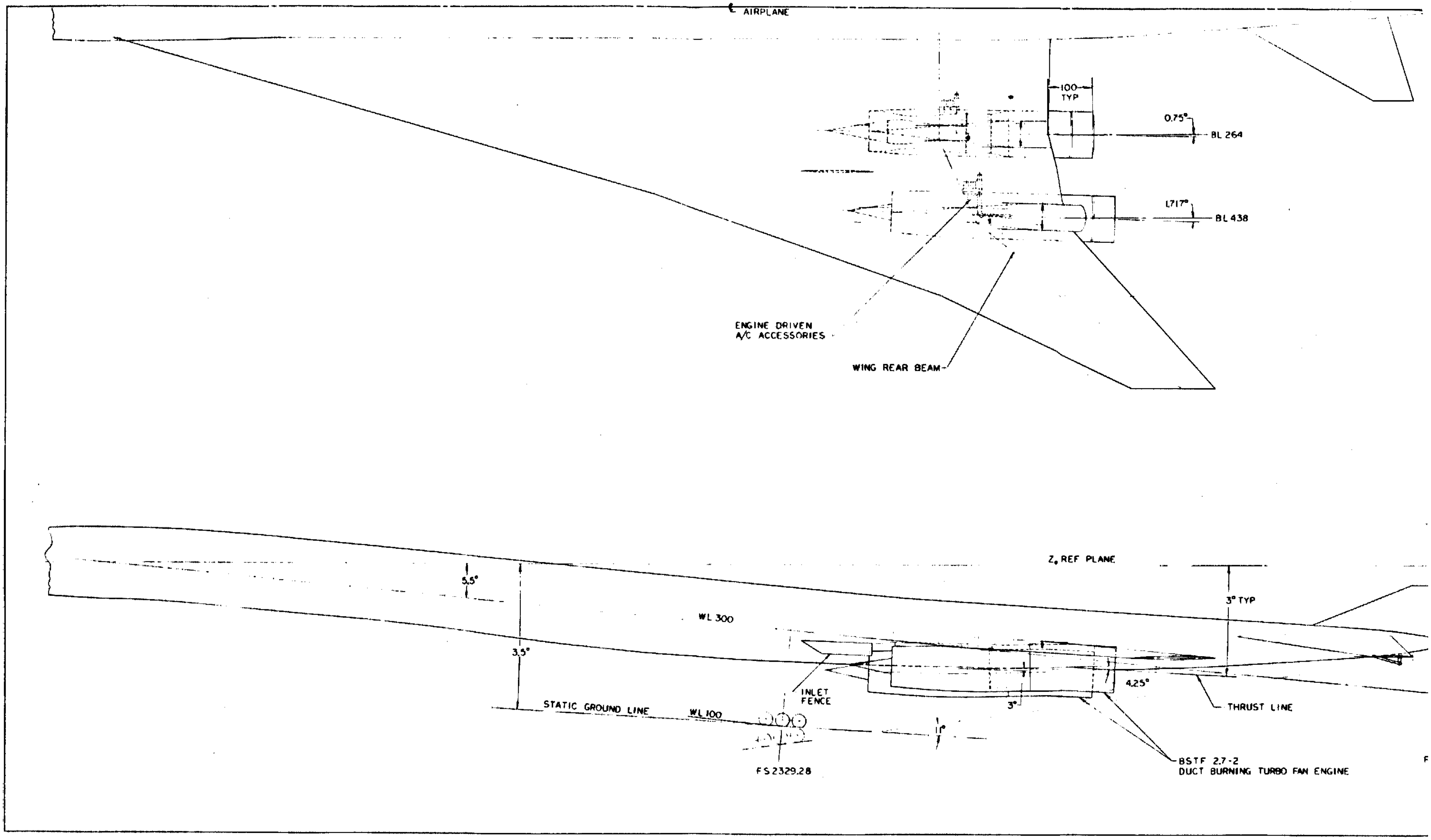
The orientation of the nacelles of the baseline configuration concept (Section 2) were derived from unpublished data of wind tunnel tests at the NASA Langley Research Center with their spanwise location determined from the NASA configuration data deck (identified as 733-336C Follow-on, April 1973). The engine characteristics for the selected duct-burning turbofan engine, designated BSTF 2.7-2, were obtained from Reference 1.

The aforementioned data were integrated into the baseline configuration concept to define the propulsion installation shown on Figure 19-1. Small longitudinal and spanwise movements from the baseline position were explored and sensitivities determined to aid in establishing the best aft mounted-underwing installation for the propulsion packages.

DESIGN INTEGRATION

The effect of engine location and size on the overall airplane structural, aerodynamic and performance characteristics were explored to establish a nacelle constraint envelope for the placement of four underwing nacelles on the arrow-wing configuration transport.

The duct-burning turbofan engine shown in Figure 19-2 was selected for the study. The engine, designated the BSTF 2.7-2, has a bypass ratio of 3.26 and a fan pressure ratio of 3.0. The scale-one engine has an uninstalled sea level static thrust of 78,000 pounds, a maximum diameter of 90 inches and an overall length of 255 inches. Other pertinent data are found in Engine Recommendations of Section 2. Table 19-1 gives the propulsion system parameters for the BSTF 2.7-2 engine.


℄ AIRPLANE
100 TYP
0.75°
BL 264
1.717°
BL 438
ENGINE DRIVEN A/C ACCESSORIES
WING REAR BEAM
Z, REF PLANE
3° TYP
5.5°
WL 300
3.5°
4.25°
3°
1°
INLET FENCE
THRUST LINE
STATIC GROUND LINE
WL 100
FS 2329.28
BSTF 2.7-2 DUCT BURNING TURBO FAN ENGINE

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FOLDOUT FRAME 2

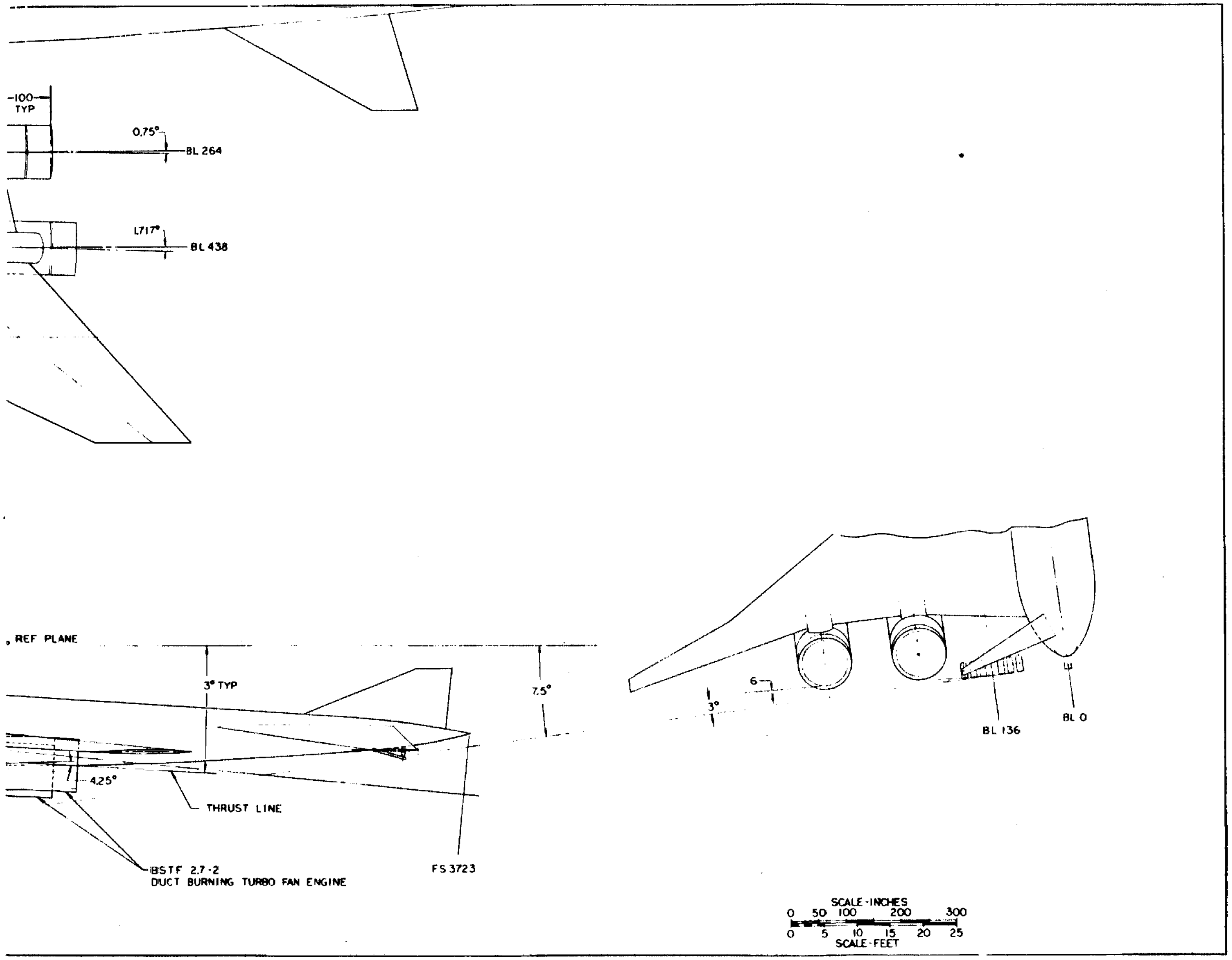
-100-
TYP
0.75°
BL 264
1.717°
BL 438
REF PLANE
3° TYP
7.5°
4.25°
THRUST LINE
BSTF 2.7-2
DUCT BURNING TURBO FAN ENGINE
FS 3723
6
3°
BL 136
BL 0
SCALE-INCHES
0 50 100 200 300
0 5 10 15 20 25
SCALE-FEET

Figure 19-1. Power Plant Configuration - AST BSTF2.7-2 Duct Burning Turbofan

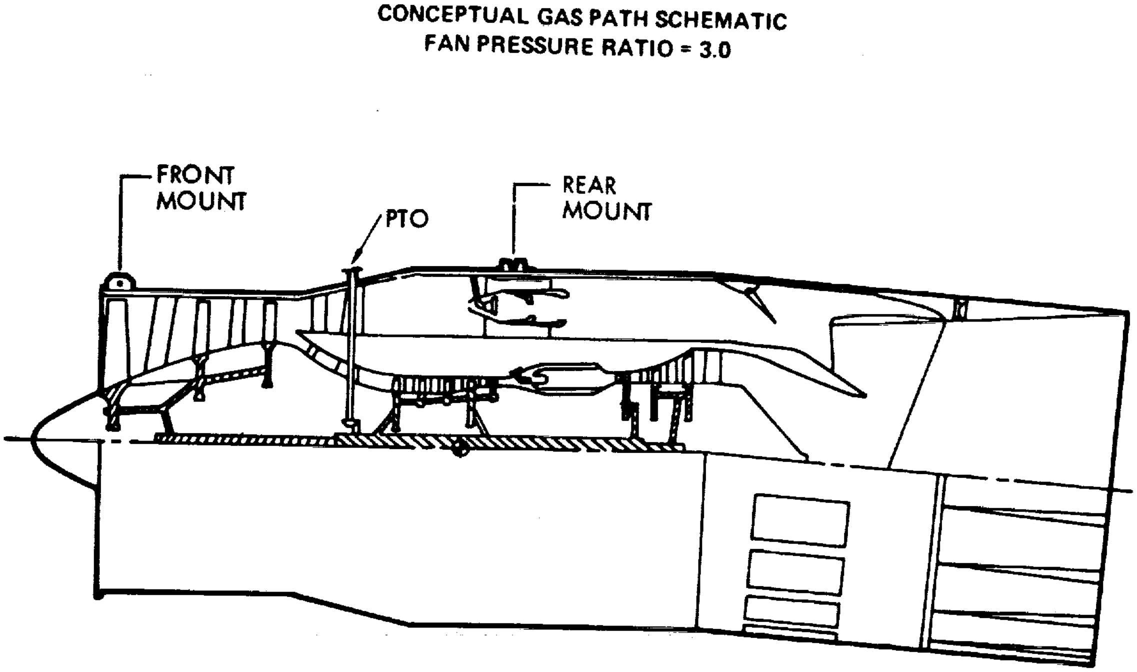
CONCEPTUAL GAS PATH SCHEMATIC
FAN PRESSURE RATIO = 3.0
FRONT MOUNT
PTO
REAR MOUNT

Figure 19-2. Duct Burning Turbofan Engine - Mach 2.7

TABLE 2-2. PROPULSION SYSTEM PARAMETERS

Engine:	BSTF 2.7-2 duct burning turbofan	
Number of engines:	4	
Noise suppression:	FAR 36-5	
Inlet/nozzle:	Axisymmetric/variable convergent-divergent	
Thrust/weight — (lift off):	0.36	
Lift of Speed:	Mach 0.30	
Scale Factor:	1.0 (Ref.)	1.147
Net thrust, lb. (A)	78,000	89,466
Engine weight, lb. (B)	11,143	12,781
ACAP, ft^2	33.1	38.0
DMAX, in.	90	96.4
DCOMP, in.	79.4	85.0
DNOZ, in.	90	96.4
LENG, in.	255	267.5
LINLET, in.	189.3	203.9
Study Application	Task I	Task II

(A) SLS, Max. Power, uninstalled
(B) Includes reverser and suppressor

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The layout of the preliminary nacelle with an axisymmetric mixed compression inlet and the BSTF 2.7-2 duct burning turbofan engine is shown in Figure 19-3. The inlet is as described in Boeing Report FA-SS-72-50, dated April 1972, scaled to account for the differences in engine face diameter and airflow. Supersonic diffuser lines are maintained as a function of Mach number and subsonic diffuse divergence is kept unchanged. The nozzle is rotated down 4-degrees 15-minutes relative to the engine center line to permit proper orientation of nacelles relative to the wing.

The propulsion installation of Figure 19-1 are toed in 0.75-degree and 1.717-degree for the inboard (BL 264) and outboard (BL 438) engines, respectively. The inlet is aligned with the underwing flow field and the nozzle tilted 3-degrees down relative to the Z_0 reference line. The tilt prevents a loss in airplane lift/drag ratio as was determined in previous supersonic transport studies. The engine dimensional data used corresponds to the BSTF 2.7-2/(scale 1.147) duct-burning turbofan which has a maximum diameter of 96.4 inches. As shown in the figure, the rotation of the nozzle to prevent an L/D loss, the increased engine diameter, the nozzle clearance for a 3-degree roll at touch down requires an increase of 19 inches in the landing gear length from that shown for the Task I-Baseline Configuration concept. The engine mounts are located aft of the wing rear beam, and are supported from beams which extend aft (cantilevered) of the structural box. Engine accessories are arranged around the forward portion of the engine. The hydraulic pumps, constant speed driven generator, and engine power takeoff are located aft of the

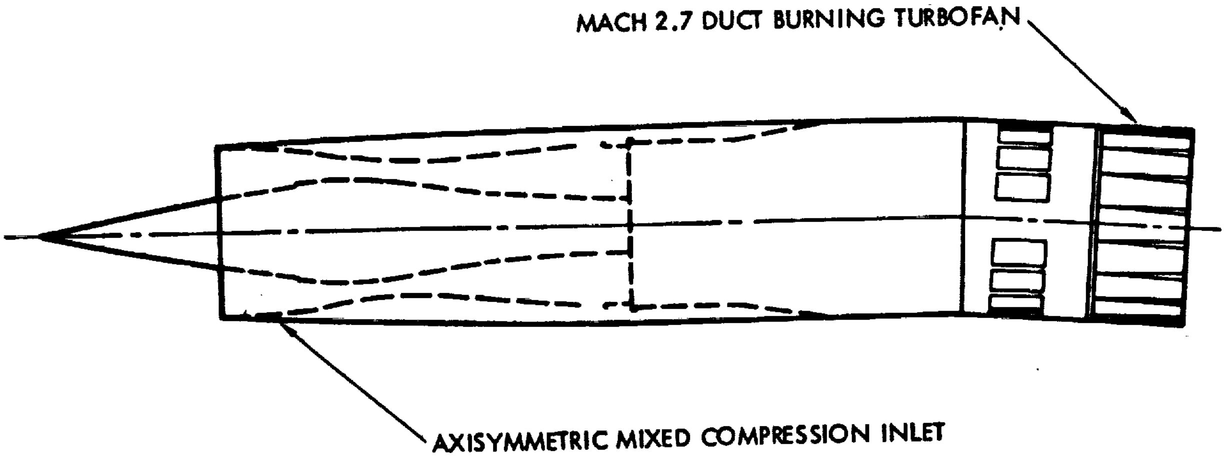
MACH 2.7 DUCT BURNING TURBOFAN
AXISYMMETRIC MIXED COMPRESSION INLET

Figure 19-3. Preliminary Nacelle

wing box, as indicated on the figure. Further study indicates that these units should be moved forward of the rear beam, with an access panel located in the upper surface of the wing box. The area aft of the rear beam is congested with controls for the flaperons, flaps and spoilers. The engine driven compressor for the airplane environmental control system is located on the engine lower centerline and requires a faired bump in the lower nacelle (not shown on drawing). Inlet boundary layer diverter heights are 15 inches on the inboard nacelle and 8 inches on the outboard nacelle. The outboard inlet is aft of the inboard inlet and approximately one diameter away. Mutual interference will cause an unstart on the inboard inlet to unstart the outboard inlet. To prevent this an inlet fence is required between nacelles as shown.

Engine Forward Movement

Factors influencing longitudinal limits on nacelle placement include wing leading edge shock wave, flutter, sonic fatigue, thermal stresses, wave drag, airplane balance, airplane tail-down and roll angle, landing gear length, and inlet mutual interference.

The arrow-wing configuration in this study has a subsonic wing leading edge. This permits the underwing nacelles to be moved forward until the inlet lip is ahead of the wing leading edge and still remain in a favorable interference field. Practical limitations at the wing trailing edge reduced the potential for the forward movement of the nacelles to approximately 100 inches.

Engine longitudinal placement effect on flutter speed as presented in the Sensitivity Studies section (Vibration and Flutter, Section 9) indicated an increase in the bending and torsion mode flutter speed with a forward engine movement (Figure 19-4). This allows for a reduction in the mass to be added to the wing tip structure, and also a mass reduction resulting from shortening of the deflection critical engine rails.

As the engine is moved forward, the effect of nozzle exhaust increases sonic fatigue and thermal stresses on the trailing edge flaps, flaperons, and wing vertical fin. The increased sonic environment is visually displayed on Figures 19-5 and 19-6. The primary mass penalty, as indicated, is on the trailing edge structure.

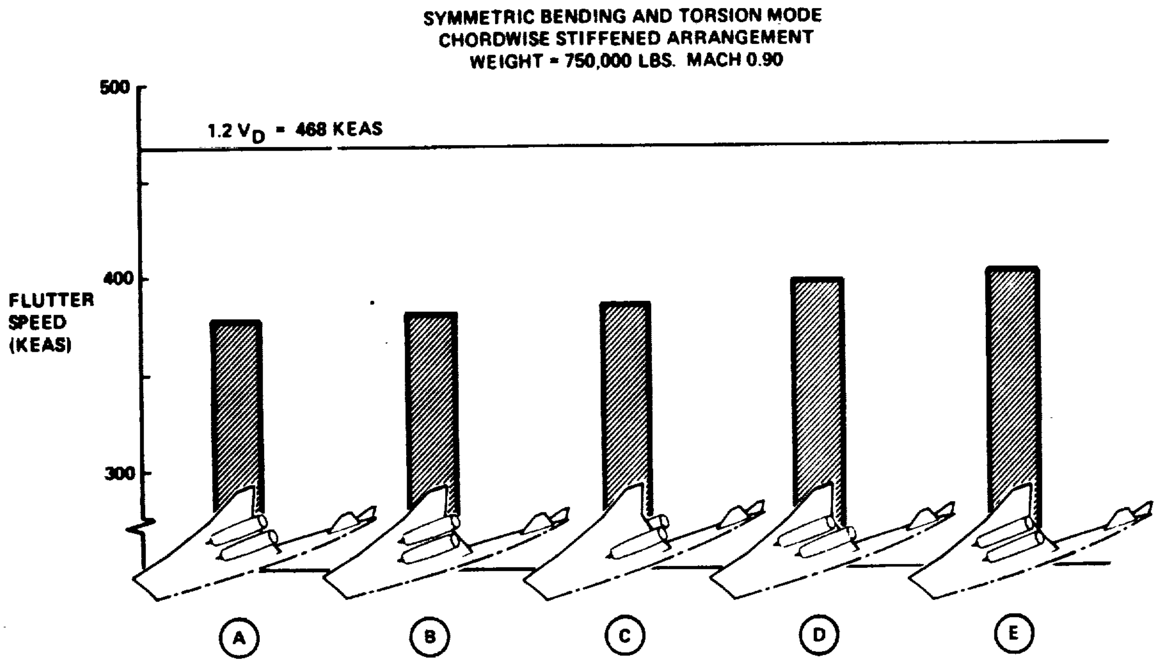

Figure 19-4. Flutter Speed Variation With Engine Placement

The forward movement of the nacelles however, permits a shorter landing gear for a given tail-down angle and roll-angle on landing. Fore and aft placement of the nacelles is further influenced by center of gravity restrictions in the balance of the airplane. Preliminary balance studies with the engine exhaust at the wing trailing edge indicates that the required balance can be achieved by shifting the fuselage relative to the wing and nacelles. The influence of an inboard inlet unstart disturbing or unstarting the outboard inlet is a function of longitudinal separation as well as spanwise and is discussed in the Propulsion System Integration section.

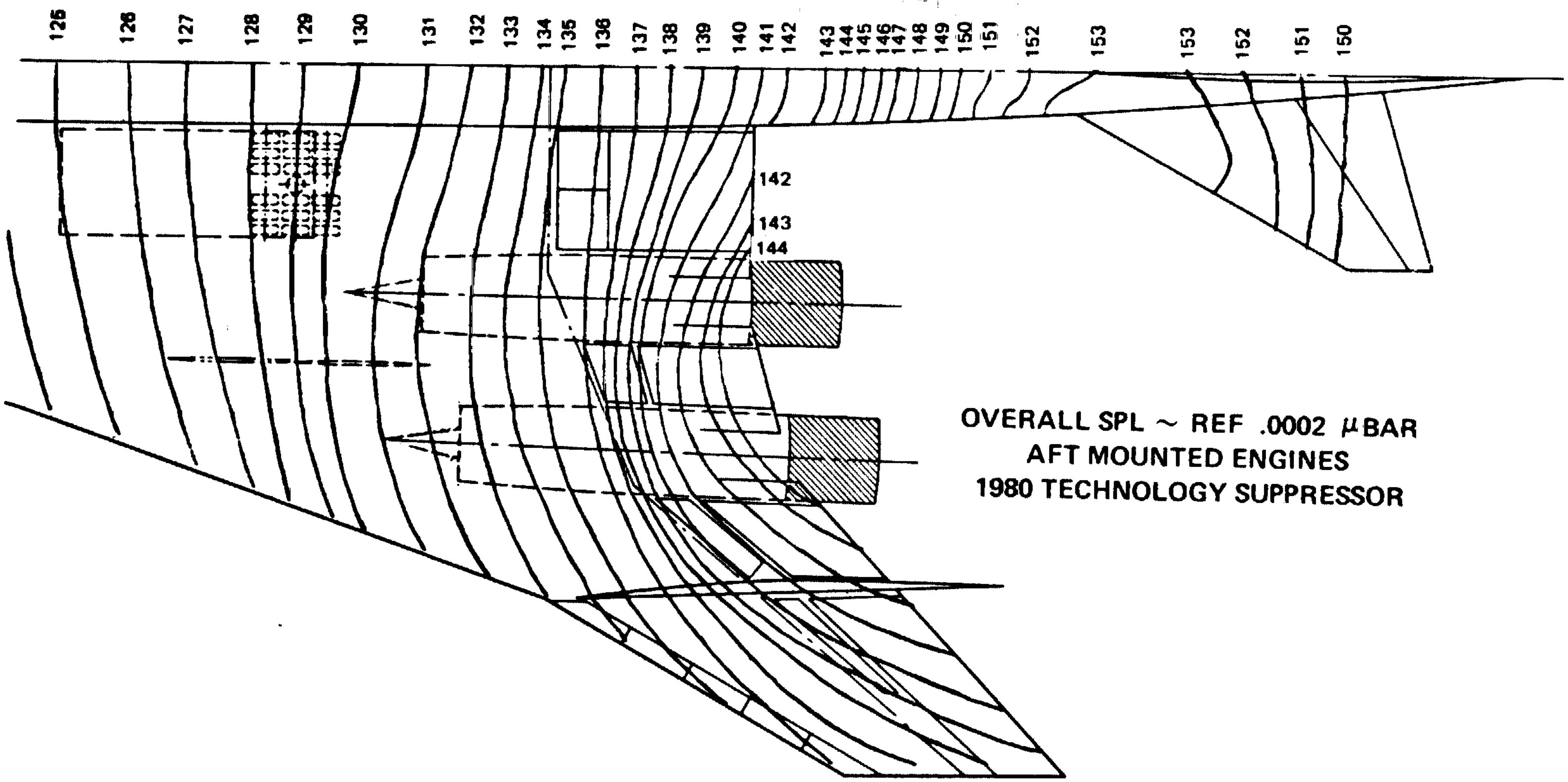

Figure 19-5. Overall Sound Pressure Level - Aft Mounted Engines

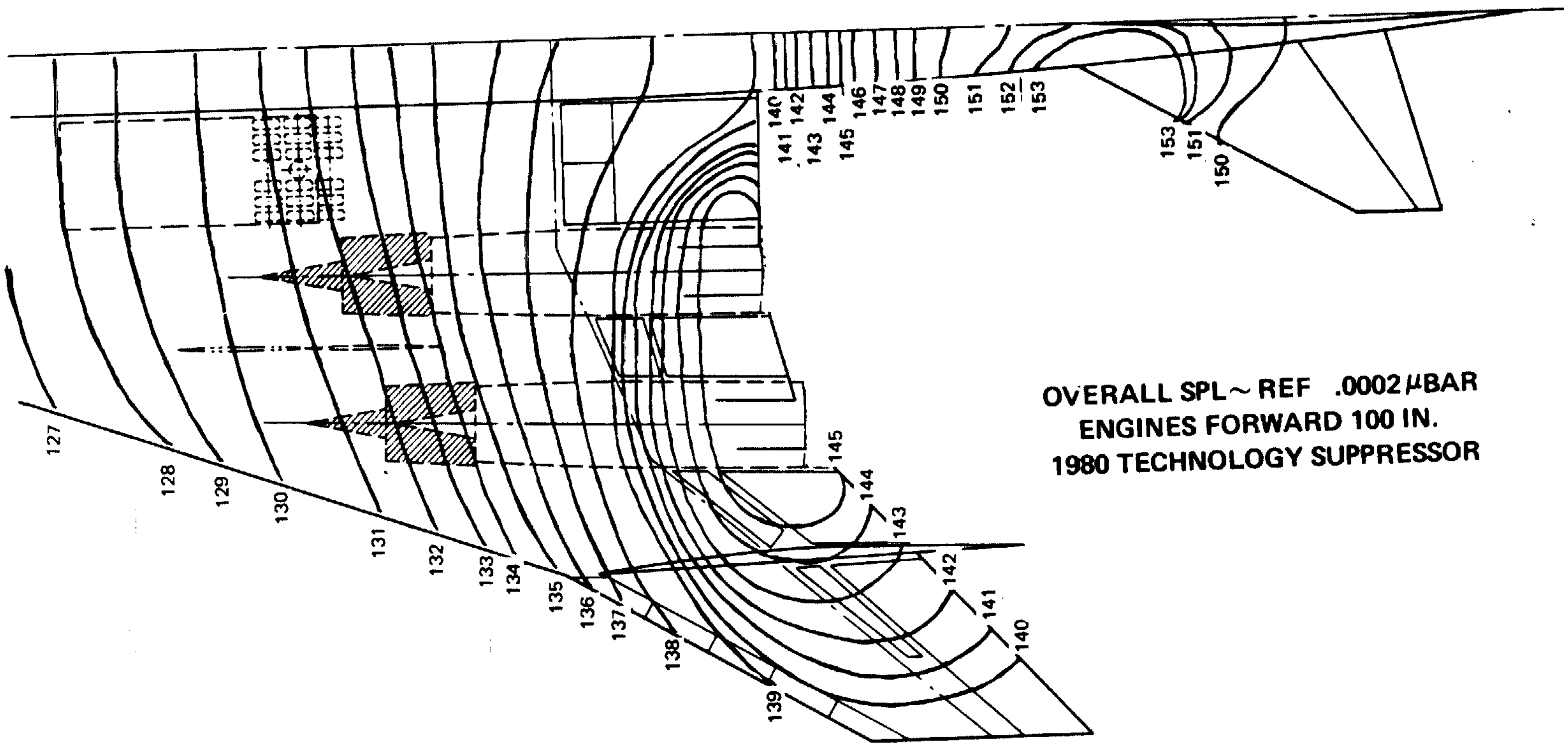

Figure 19-6. Overall Sound Pressure Level - Engines Forward

Spanwise Engine Movement

The limits for spanwise location of the nacelles were influenced by wing trailing edge control surfaces, inlet mutual interference, flutter, and main landing gear location. Of these factors, the trailing edge control surfaces were the single most critical factor in spanwise location of nacelles. With the engine size shown on Figure 19-1, lift and roll control are adequate, but there is no margin for moving the inboard-nacelle inboard. Spanwise movement would require a larger wing area or change in aspect ratio and span. In addition, moving the engines inboard and forward must consider clearance with landing gear doors and the main landing gear strut. Moving the outboard engine outboard increases roll control and incremental lift from the flap, but increases the flutter problem and requires a larger all moving vertical tail. Since movement of the two nacelles away from each other was restricted by the above factors, it was not possible to separate the inlets by one and a half diameters and line them up fore and aft. Therefore, the inlet fence must be retained to prevent an inboard inlet unstart from upsetting the outboard inlet.

Engine Size

The effects of engine size on the overall airplane performance were evaluated by arbitrarily increasing the size and weight of the BSTF 2.7-2/1.147 duct-burning turbofan engine. This increase was based on the assumption that the engine design was over-optimistic. The engine maximum diameter was increased from 96.4 inches to 107 inches, while the length remained unchanged. The engine T/W ratio was arbitrarily reduced from 7.0 to 6.0 for an increase in engine weight of 2,130 pounds per engine. The increased engine and structure weight resulted in a loss of over 200 nautical miles. The most difficult problem faced by this increased engine diameter on the arrow-wing configuration was in the wing trailing edge control surfaces. As discussed earlier, they are just adequate for the 96.4 inch diamter engine, but would be reduced to the point of insufficient lift and roll control for takeoff and landing with the larger engines. The only solution to this problem is a larger wing or increased span with resultant increased gross weight. The importance of controlling engine size, especially diameter and weight, was made very apparent in the development of the smallest, most efficient supersonic aircraft.

Constraint Envelope

The placement of four underwing engines on the arrow-wing configuration was extremely limited. As noted earlier, there was no improvement from moving the nacelles spanwise, and therefore the constraint envelope, shown in Figure 19-7, maintains the same spanwise location as the NASA card deck. Longitudinal limits were not so restrictive, and as the constraint envelope indicates, the nacelles may move from the exhaust 100 inches aft of the wing trailing edge to the exhaust at the trailing edge. Movement farther forward was possible, but increases the sonic fatigue problem. Because of the lack of benefits identified in moving the engines spanwise, the primary effort was in the determination of the best longitudinal position for the nacelles.

PROPULSION SYSTEM INTEGRATION

Boundary Layer Diverter

Boundary layer diverter height variation with nacelle longitudinal position (relative to the wing leading edge) was determined to aid in the design layout. The results shown in Figure 19-8 are independent of spanwise position and were derived from unpublished data of wind tunnel tests conducted at the NASA Langley Research Center. The diverter heights for the baseline configuration of the arrow-wing supersonic cruise aircraft are indicated on the figure.

Inlet Mutual Interaction

An important aspect of airframe/propulsion system integration is the minimization of inlet-to-inlet interaction effects. These mutual interaction effects can be particularly severe for underwing engine installations. Specifically, the disturbance associated with an inlet unstart may propagate through shock-boundary layer interaction effects to an adjacent inlet. If the disturbance is sufficiently strong the adjacent inlet will unstart. In order to prevent excessive aircraft pitch, yaw, and rolling moments, an adjacent inlet unstart should be prevented.

An analysis of experimental data (Reference 2) for inlet mutual interaction effects at an inlet Mach number of 2.6 indicates that a boundary layer fence

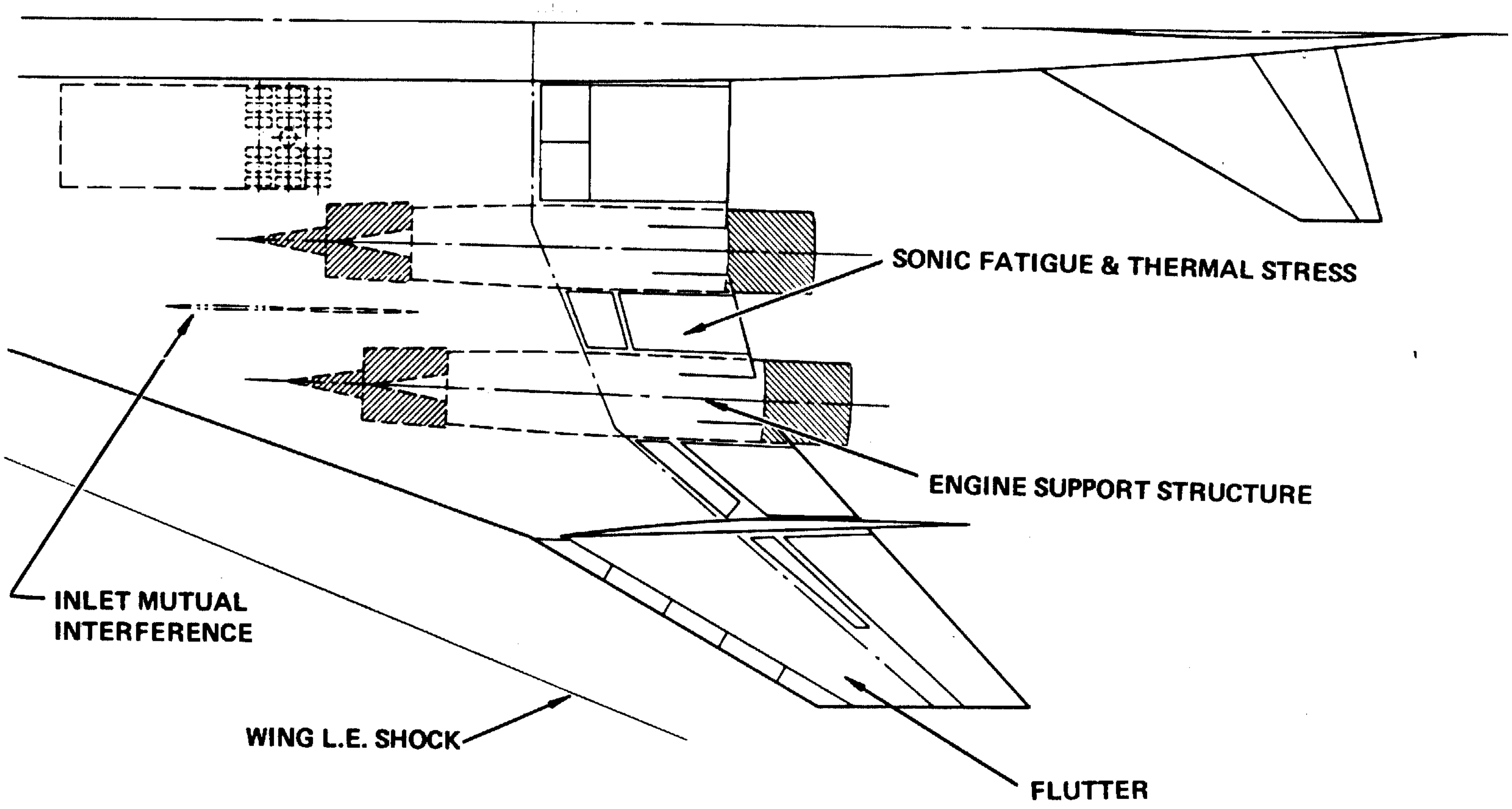
SONIC FATIGUE & THERMAL STRESS
ENGINE SUPPORT STRUCTURE
INLET MUTUAL INTERFERENCE
WING L.E. SHOCK
FLUTTER

Figure 19-7. Constraint Envelope

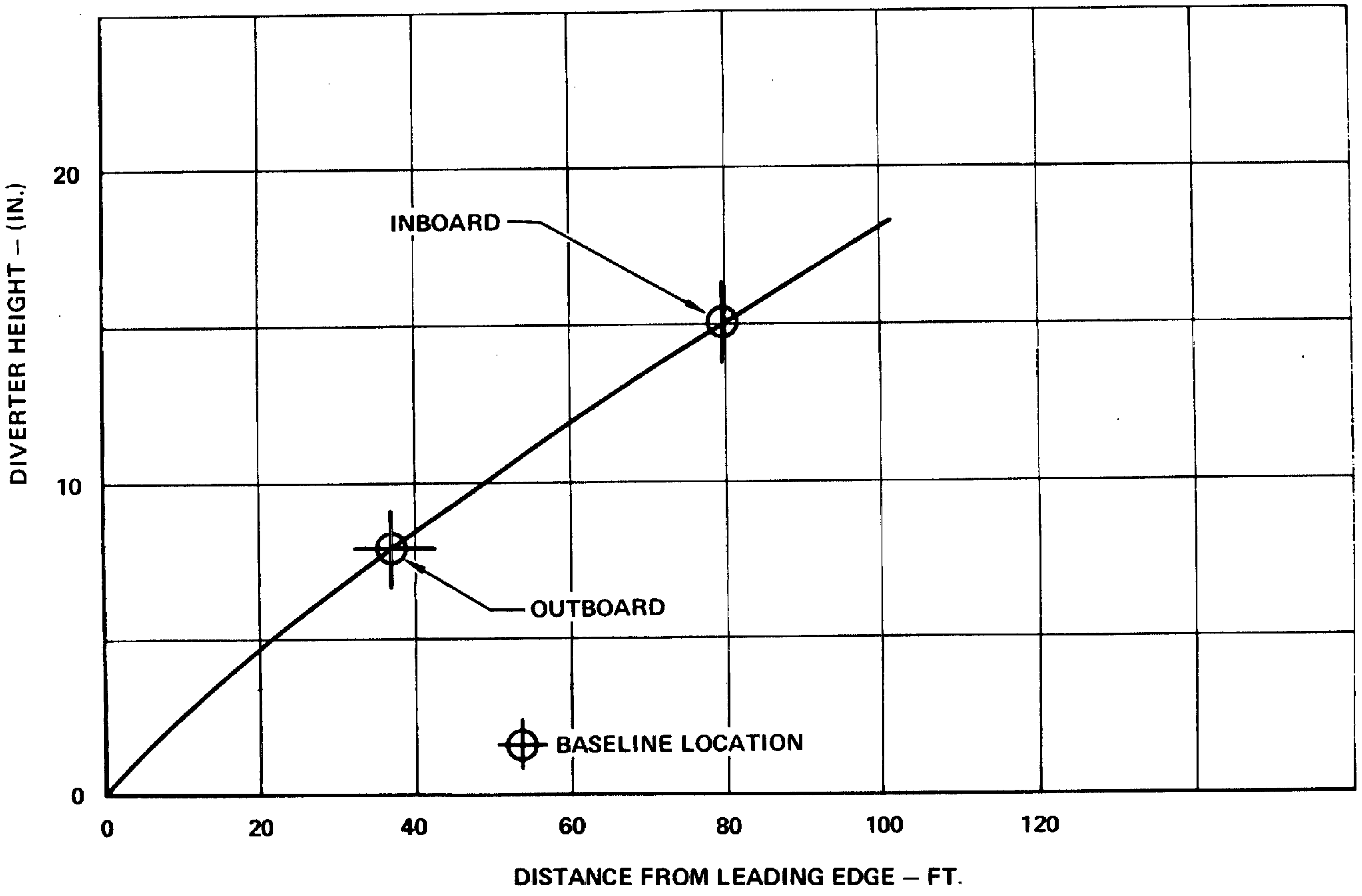

Figure 19-8. Diverter Height Variation With Nacelle Longitudinal Position

will be required to prevent an unstart of the downstream outboard inlet due to an unstart of the inboard inlet of the baseline study vehicle. Figure 19-9 shows the stable and unstable regions for operation of the outboard inlet during unstart of the inboard inlet, based on experimental data for two inlets with no wing. The results, which show the present location to be only marginally satisfactory, do not reflect the effect of the interaction between the expelled inlet shock wave and the wing boundary layer. Since the expelled inlet shock wave will propagate further forward due to separation of the wing boundary layer, the outboard inlet will most likely unstart. A boundary layer fence, similar to the one employed on the Boeing SST, was incorporated on the baseline vehicle to prevent this unstart.

An outboard inlet unstart can also be prevented by relocation of the inlet. Based on the Mach 2.6 data of Reference 3, the inlets would have to be laterally aligned and separated approximately three and one-half inlet diameters (cowl lip to cowl lip) to completely eliminate mutual interaction effects. Depending on the sensitivity of the inlet and engine to the adjacent inlet unstart disturbance, the inlets could possibly be placed closer together. However, because of the complex interactions involved, the minimum inlet separation distance would have to be determined experimentally.

As discussed in Reference 4, a lateral separation distance of one and one-half equivalent inlet diameters was found to be acceptable for the Lockheed L-2000 SST. The equivalent inlet diameter is defined as the diameter of an axisymmetric inlet which has the same capture area as the L-2000 SST two-dimensional inlet. This result was determined by tests of a scale model inlet in the presence of a simulated wing boundary layer and a simulated adjacent inlet unstart disturbance. The unstart disturbance was simulated by air injection through slots into the wing boundary layer, controlled to yield the same wing pressure distribution as previously determined from small scale aircraft/inlet model tests. Since two-dimensional inlets were employed in the above tests, the inlet stability results may not be completely applicable to axisymmetric inlet installations since the shock wave boundary layer interactions differ from one to the other.

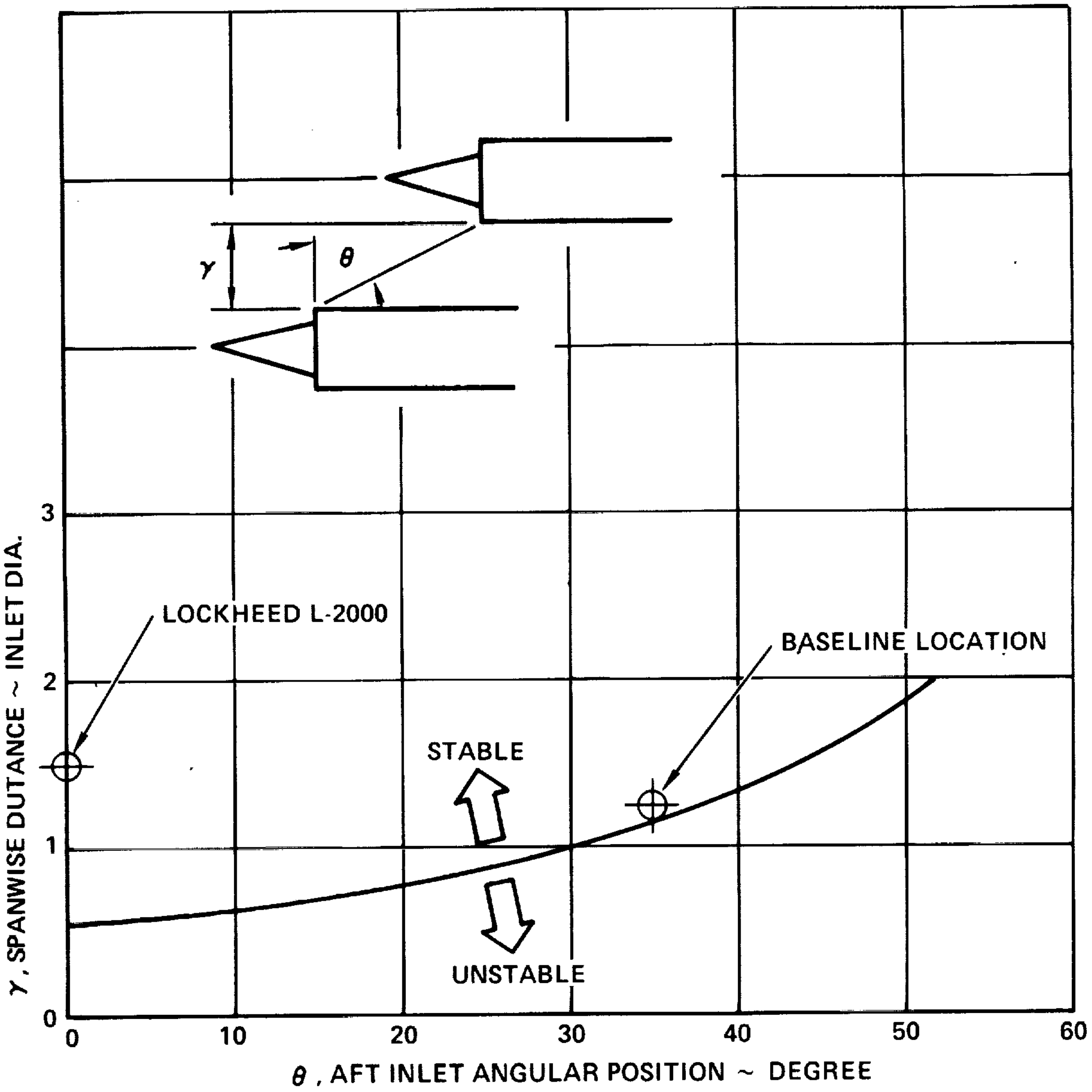

Figure 19-9. Stable Operating Location Of Aft Inlet

AERODYNAMIC CHARACTERISTICS

The analyses to determine the aerodynamic effects of moving the engine nacelles from their baseline location (Figure 19-1) were performed. The range of potential locations was limited to spanwise positions along the lower surface of the wing trailing edge inboard of the wing vertical fin. Spanwise shifts of either nacelle cause changes in the size and location of the wing trailing edge flap panels. Also, the yawing moment due to the thrust of the outboard engine is changed.

Control Surface

The panels affected by the spanwise shifts studied are F_1, F_2, and F_3 whose relative positions are shown in the sketch. Flap panel F_1 is used only as a lift flap while F_2 and F_3 are flaperons providing both incremental lift and roll control. Figure 19-10 shows the incremental lift coefficient due to all three flap panels deflected 20-degrees. Both nacelles are shifted inboard and outboard from their baseline positions keeping the separation between them constant at the baseline value. This provides a trade off between panel F_1 and panel F_3. Because the effectiveness of a flap is dependent in part on the area of wing in front of it, the addition of span to the inboard panel F_1 at the expense of outboard F_3, increases the lift added by about ten percent. Similarly, a decrease of F_1 and increase of F_3 causes a net loss in flap lift effectiveness.

Figure 19-11 shows the effect of a trade between F_2 and F_3 while F_1 was held constant. The airplane trimmed lift coefficient for conditions approximating second segment climb is shown as a function of outboard nacelle location in Figure 19-12.

Variation in the incremental lift potential of flaperon panels F_2 and F_3 coupled with the shift in panel center of pressure effects the roll control power available at Mach numbers up to 0.8. Rolling moment coefficient from F_2 and F_3 at full asymmetric deflection is shown in Figure 19-13. The individual contributions and their sum are shown as a function of outboard nacelle spanwise location.

A shift of the outboard engine inboard or outboard from the baseline position will change the yawing moment due to an inoperative engine. The all moving

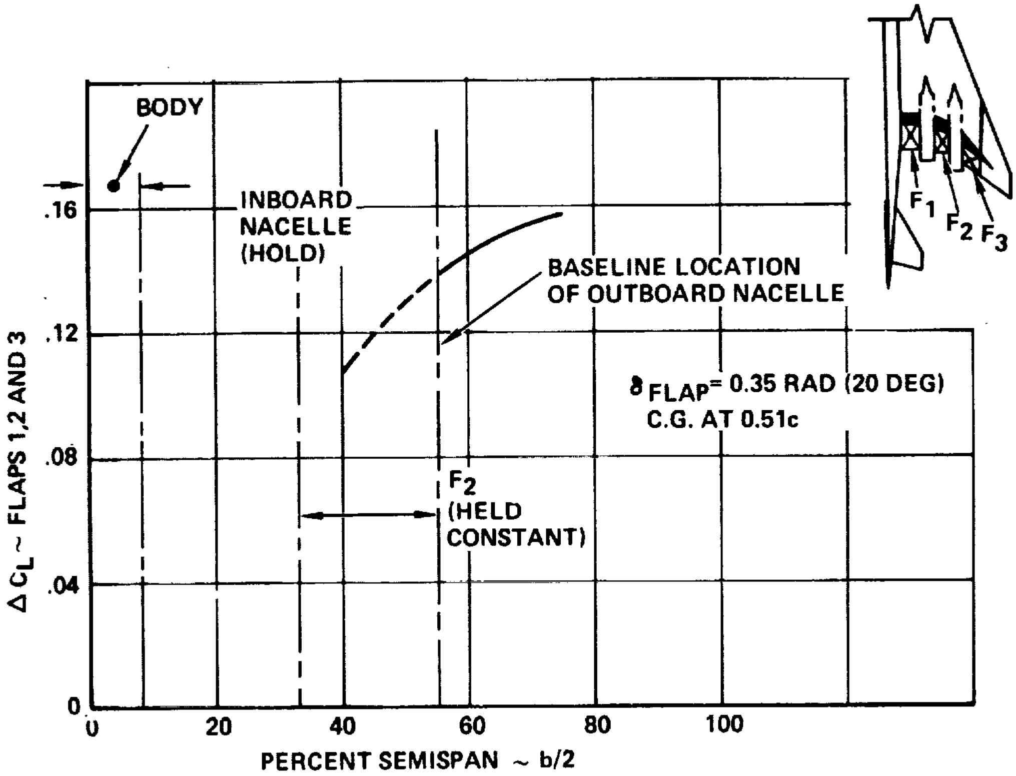

Figure 19-10. Incremental Lift Due To Inboard And Outboard Engine Spanwise Shift - Flaps at 20 Degrees

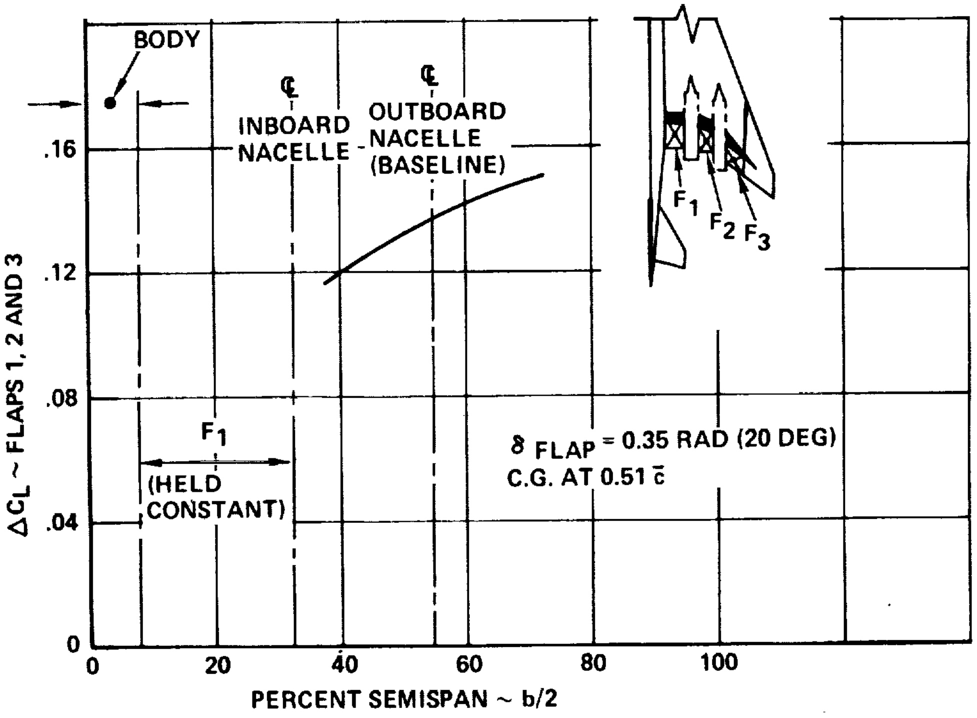

Figure 19-11. Incremental Lift Due To Outboard Engine Spanwise Shift - Flaps at 20 Degrees

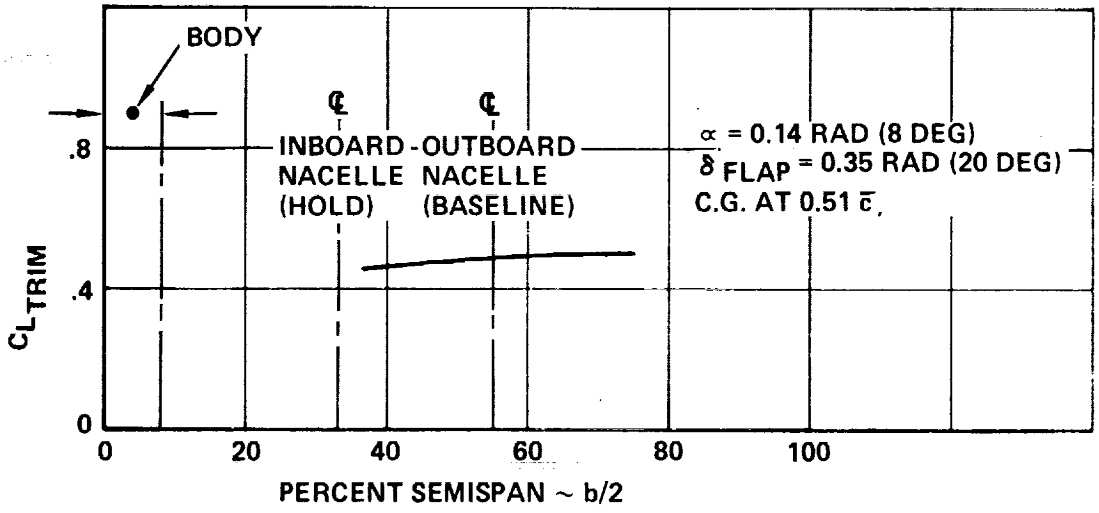

Figure 19-12. Trimmed Lift As A Function Of Outboard Engine Spanwise Location - Flaps At 20 Degrees

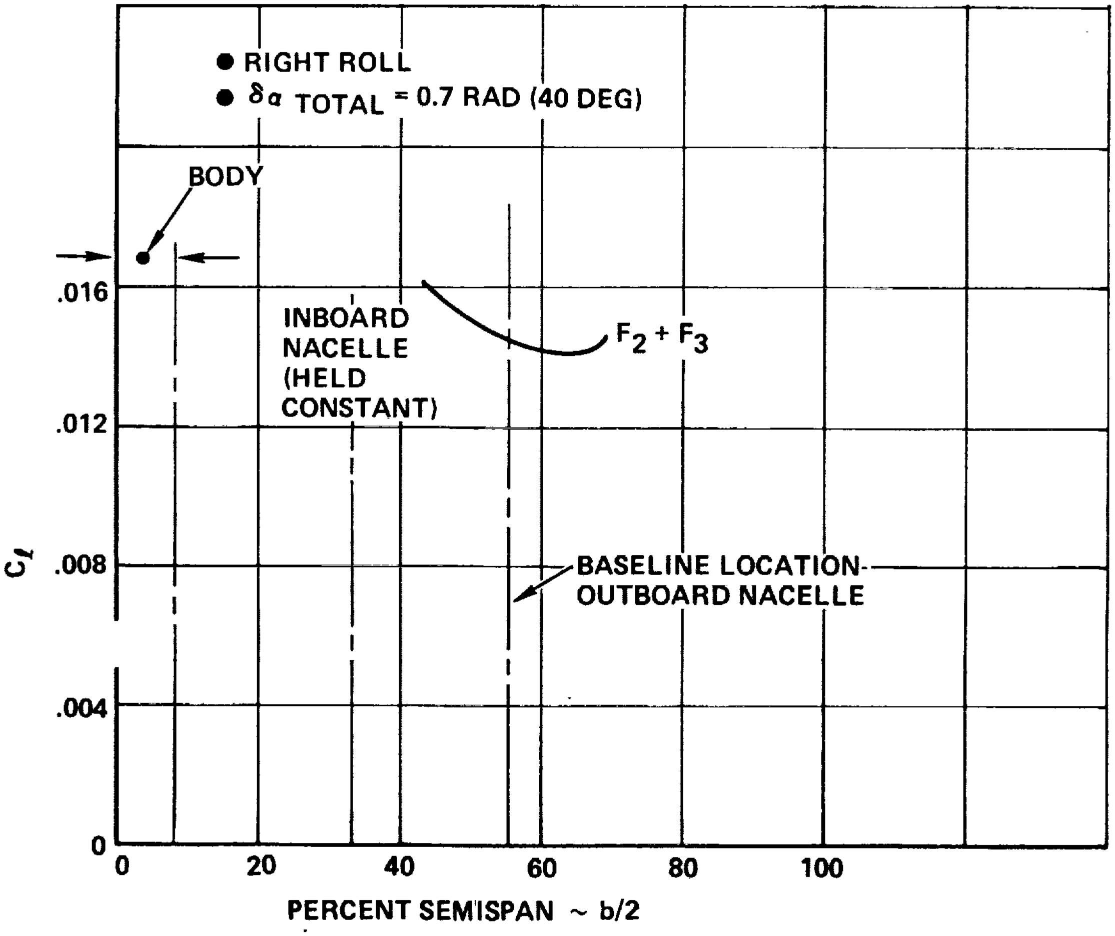

Figure 19-13. Effect Of Outboard Engine Spanwise Location on Rolling Moment Due to Flaperons

centerline vertical tail is sized to control this thrust asymmetry at low speed as required for acceptable performance. The dependence of this tail area on outboard engine position is shown for constant minimum control speed in Figure 19-14.

Performance

Fore and aft position of the engine nacelles does not modify relative flap size or vertical tail area and hence does not effect low speed aerodynamics. It does, however, effect wave drag at supersonic speeds. If both inboard and outboard nacelles are shifted 100 inches forward the wave drag is increased by approximately one count (0.0001) at cruise. One count of drag has the same effect on performance as adding 2500 pounds of empty weight or reducing range 31 nautical miles. The mission maximum lift to drag ratios for the baseline airplane and for the airplane with nacelles forward are shown in Figure 19-15.

STRUCTURAL CHARACTERISTICS

The structural effects of longitudinal and spanwise movements from the baseline position were identified with the change in flutter characteristics, sonic fatigue requirements, and mass and balance as discussed in the Design Integration section.

Flutter Characteristics

The flutter optimization results and engine placement investigation reported in Section 10, Vibration and Flutter, provided an insight into the structural mass trends with engine longitudinal position. The results of the study (Figure 19-4) indicates an increase in the flutter speed for the bending and torsion mode from 379 KEAS to 401 KEAS or an increase of 22 KEAS. This increase in flutter speed allows for a reduction of approximately 500 pounds of structure from the wing tip region. In addition, shortening of the engine support beams by 100 inches each permits a mass savings of approximately 450 pounds. Thus, the net potential savings is 950 pounds per aircraft.

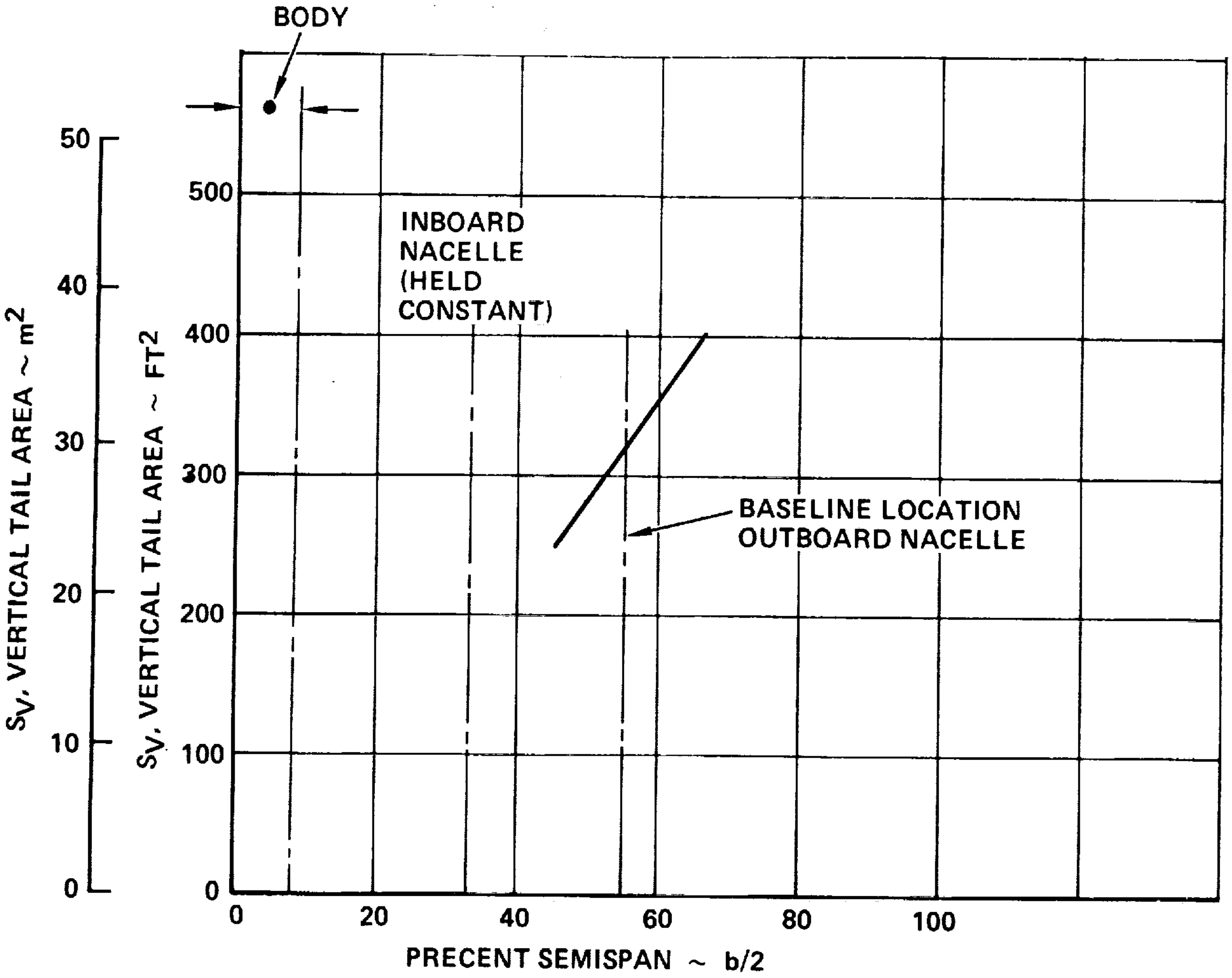

Figure 19-14. Vertical Tail Required Due To Spanwise Shift Of Outboard Engine

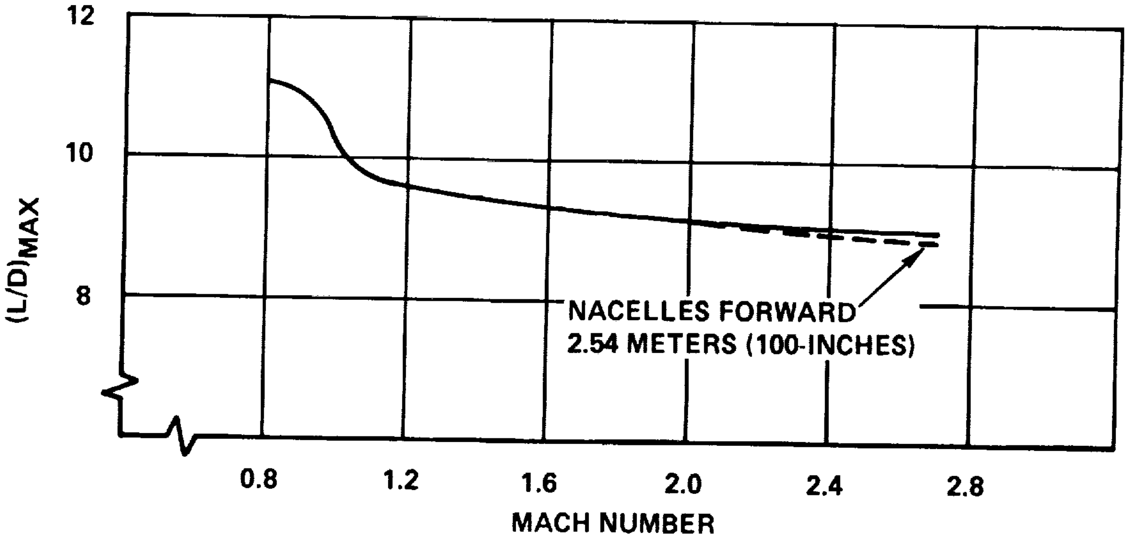

Figure 19-15. Effect Of Mission Lift/Drag Ratio On Nacelle Forward Shift

Sonic Fatigue

The change in the acoustic environment on the trailing edge structure was assessed for a fixed geometry honeycomb core sandwich structure. The increase in thickness of titanium face sheet for the flap, flaperon and aileron resulted in a mass increment of 50 pounds.

RESULTS

The propulsion-airframe integration study concentrated on the benefits of the exhaust being 100 inches aft of the wing trailing edge versus the exhaust at the trailing edge as shown in Figure 19-16. By moving the engines to the forward position, the mass added to the outer wing to prevent symmetric flutter was reduced by 500 pounds. The support structure for the engines (beams cantilevered off the rear beam) could also be reduced 450 pounds. The landing gear could be shortened 15 inches for a saving of 1100 pounds. The flaps, flaperons, and fixed vertical tails are designed for higher sonic fatigue levels, which would add 50 pounds. The net mass reduction would be approximately 2000 pounds for a 42 nautical mile increase in range as shown in Table 19-2. Against this is a reduction of 31 nautical miles in range due to an increase in wave drag when the nacelle moves forward. While the results appear inconclusive due to their closeness, it would appear that the engines should be maintained at the baseline location for the Task II design effort.

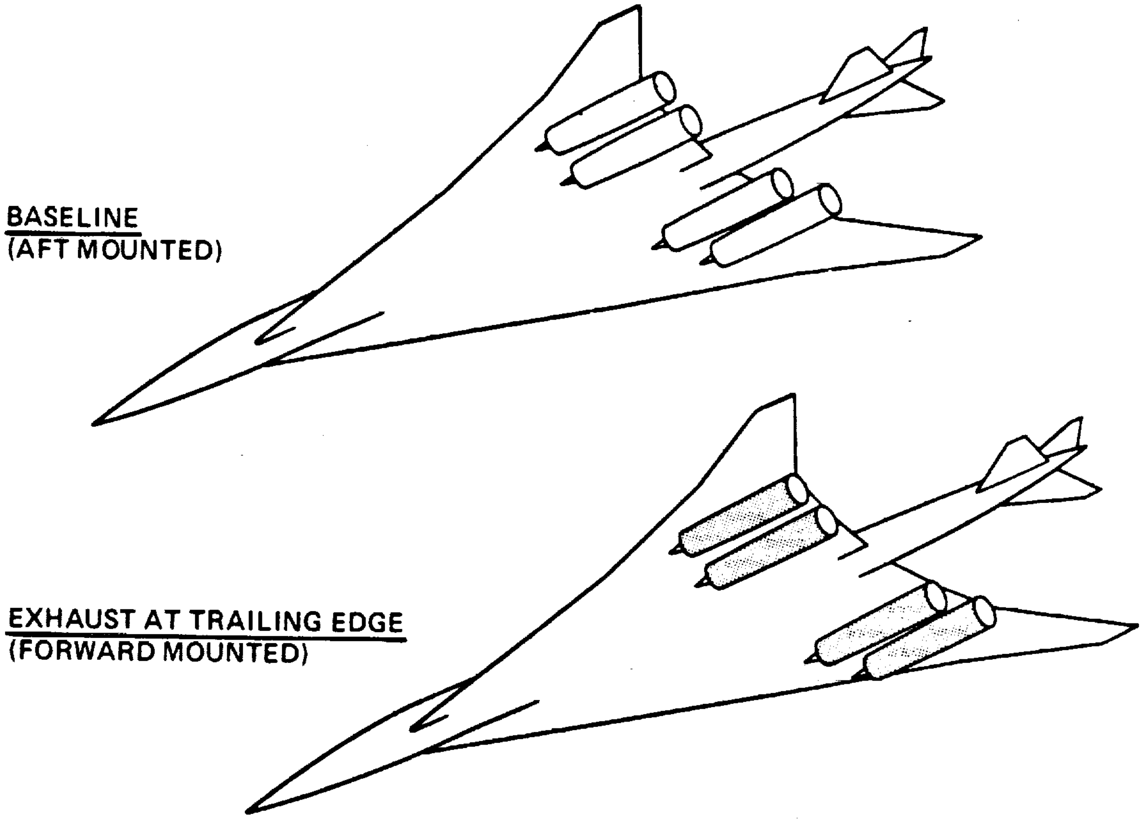
BASELINE
(AFT MOUNTED)
EXHAUST AT TRAILING EDGE
(FORWARD MOUNTED)

Figure 19-16. Candidate Engine Location

Table 19-2. Propulsion - Airframe Integration Results

WEIGHT CHANGE -	
• FLUTTER	-500 LB
• ENGINE SUPPORT BEAMS	-450 LB
• LANDING GEAR LENGTH	-1,100 LB
• INCREASED SONIC FATIGUE	+50 LB
	-2,000 LB
RESULTS -	
• WEIGHT SAVING	+42 N.MI.
• WAVE DRAG PENALTY	-31 N.MI.
• NET	+11 N.MI.
CONCLUSION -	
• MAINTAIN BASELINE ENGINE LOCATION	

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SECTION 20

ADVANCED TECHNOLOGY ASSESSMENT

by

I. F. Sakata, R. N. Jensen and A. R. Holland

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LIST OF SYMBOLS AND NOTATIONS

E_1	Young's modulus of laminae parallel to filament direction (lbs/in^2)
E_2	Young's modulus of laminae transverse to filament direction (lbs/in^2)
E_{xx}	Young's modulus of laminate along X reference axis (parallel to 0 direction) - (lbs/in^2)
E_{yy}	Young's modulus of laminate along Y reference axis (parallel to 90° direction) - (lbs/in^2)
G_{12}	Shear modulus of lamina in the lamina reference axis, parallel to filaments (lbs/in^2)
G_{xy}	Shear modulus of laminate in the X-Y reference axis (lbs/in^2)
n_z	Vertical inertia load factor - inertia force parallel to the airplane vertical reference axis divided by the weight (up is positive)
N_x, N_y, N_{xy}	Inplane stress resultants (lb/in) acting in the X- and Y- direction and the X-Y plane
V_e	Equivalent airspeed (keas)
α_1	Linear coefficient of expansion in the filament direction (in/in/F)
α_2	Linear coefficient of expansion transverse to the filament direction (in/in/F)
α_B	Buckling coefficient (see text)
γ_{12}	Shear strain of the lamina in the lamina reference axis, parallel to filaments (in/in)
$\epsilon_1(T)$	Ultimate tensile strain allowable in the filament direction (in/in)
$\epsilon_2(T)$	Ultimate tensile strain allowable transverse to the filament direction (in/in)
$\epsilon_1(C)$	Ultimate compressive strain allowable in the filament direction (in/in)
$\epsilon_2(C)$	Ultimate Compressive strain allowable transverse to the filament direction (in/in)

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ν_{12}	Major Poisson's ratio of laminae relating contraction in the transverse direction due to extension in the filament direction.
ν_{21}	Minor Poisson's ratio of laminate relating contraction in the filament direction due to extension in the transverse direction
ρ	density (lbs/in^3)

SECTION 20

ADVANCED TECHNOLOGY ASSESSMENT

INTRODUCTION

The results of the Advanced Technology Assessment, which explored an aggressive application of composite materials and fabrication technology for wing and fuselage structure of a Mach 2.7 arrow-wing supersonic cruise aircraft, are presented in this section.

The analysis of selected wing and fuselage point design regions, considering the various structural arrangements, design concepts, and corresponding internal loads and flexibilities, were made by the Lockheed-Georgia Company. Weight reduction factors for secondary and other structural components were obtained from the results of the Advanced Technology Transport (ATT) studies of Reference 1 and were applied as appropriate to determine weights for an aircraft with a fixed takeoff weight of 750,000 pounds and an aircraft resized to provide a constant payload-range performance.

ADVANCED TECHNOLOGY SELECTION

Studies of advanced technology application to future transport performance and economics identified major technological advances that could reasonably be available during the 1980-1990 time period. The resulting trends indicated conclusively that the highest structural mass payoff was in the area of materials technology or more specifically, composite material system application. Furthermore, the most significant structural mass reduction resulted from resizing the airplane to reflect the lower structural weight achieved through advanced materials application.

Reference 1 presents the results of a study of the application of advanced technologies to long-range transport aircraft conducted by the Lockheed-Georgia Company. Trade studies were conducted to determine the effects of variation in technology advancements in the areas of (1) aerodynamics, (2) structures and materials, and (3) active controls systems. The most significant benefit was obtained by the application of advanced materials as shown by marked improvements on airplane performance and economics. For example, for a 50-percent utilization of advanced materials, the takeoff gross weight was reduced by about 30-percent and the ROI is increased by about 35-percent. Above 50-percent utilization of advanced materials, however, the DOC increased and ROI decreased, because of the relatively inefficient application of advanced materials in this region.

Reference 2 presents the impact of advanced technologies on supersonic transport aircraft. Weight savings predictions based on the use of composites, new structural concepts and active controls were developed. The benefit of weight reduction was amplified by the growth factor for this class of aircraft (Reference Section 17); consequently, a pound saved in structure results in gross weight being approximately 6-pounds less for a supersonic transport maintaining the same payload-range capability. This reduction results from the smaller fuel requirement and the lighter structure associated with the reduced gross weight as follows:

Basic weight saving	1.0 lb.
Fuel reduction	2.8
Structural weight reduction	2.2
Total gross weight reduction	6.0 lb.

Although no attempt was made in the study of Reference 2 to single out individual structural improvements (i.e. materials, design concepts, active controls), a forecast was made of a 10-percent weight savings taking the improvements collectively. The impact of these technology improvements is illustrated in Figure 20-1.

Also shown on the figure are weight and performance trends resulting from the advanced technology assessment of this study (designated far-term) and the Task I results (designated near-term). The shaded band represents the potential improvements by application of advanced materials and concepts. The takeoff weight is approximately 100,000 pounds less, and the range increases 500 nautical miles.

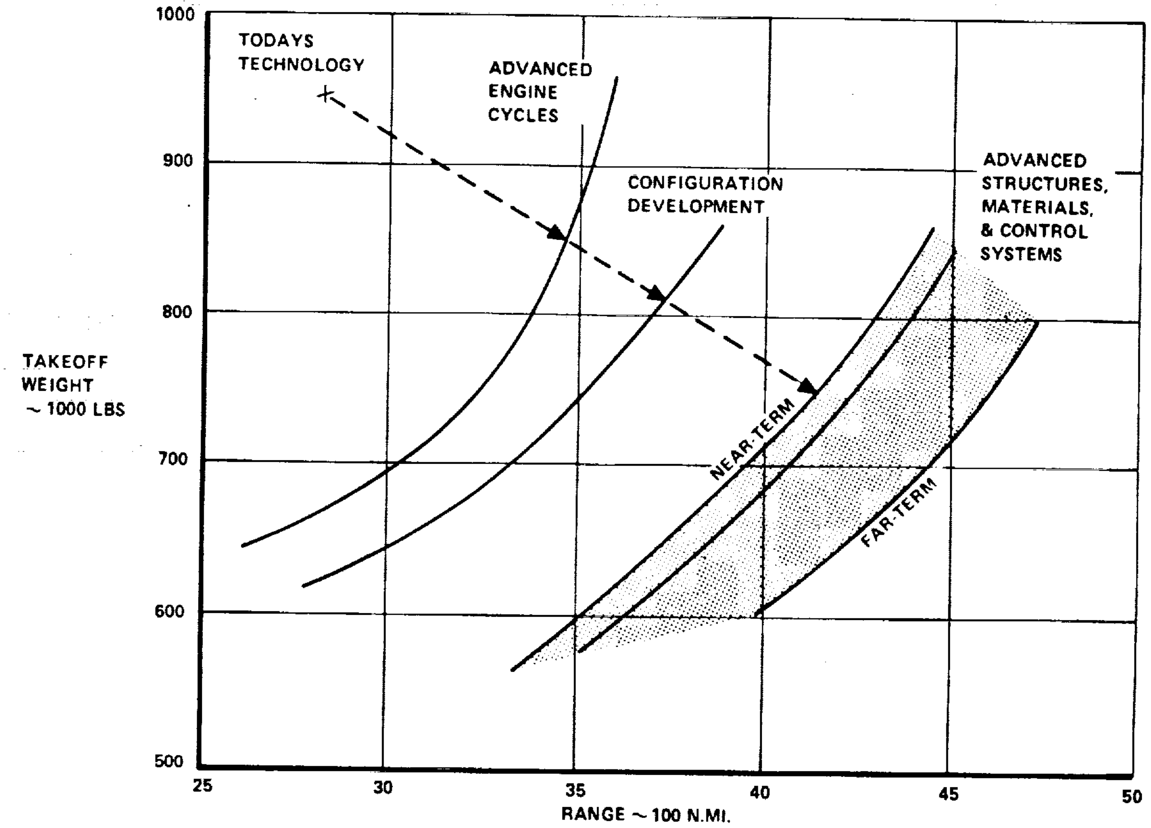


Figure 20-1. Structures, Materials, and Controls Impact

DESIGN APPROACH

Projected composite development trends predict the availability of improved stable high temperature resin systems such as thermoplastic polyimides or high temperature polyaromatics, large numerically controlled tape laying equipment, filament winding and pultrusion equipment, and larger autoclaves.

To arrive at projections for airframe structural mass for the advanced technology supersonic cruise aircraft, the results of the Task I Analytical Design Studies were used to size specific point design regions. The sizing data included the internal loads and stiffness requirements of the various airframe arrangements (i.e. chordwise stiffened, spanwise stiffened and monocoque designs). A comparison was then made with the minimum weight all-metal design to similar designs in graphite or boron composites.

Basic section weights were taken as the basis of comparison since nonoptimum factors resulting from advanced manufacturing techniques used for the near-term aircraft (e.g. welded design) assembly are offset by a bonded composite structure having approximately equal utilization of mechanical fasteners. Further, historical data for composite assemblies of the size and type considered are not available.

The use of composite materials to the extent considered in this study is dependent upon two restrictions related to the materials themselves. First, it was assumed that high service temperature capability polyimides would be available and that the material system could be processed with far less difficulty than 1974 polyimide resin systems. Processing of currently available high temperature systems would increase the complexity of assembly fabrication greatly. Second, it was premised that for the sake of economy, assembly sections were large and automatically produced. However, the time required for lamination, handling of such a large volume of material (by machine or otherwise), and the molding of part or assemblies of the sizes required offer problems which have never before been faced.

Materials

Design Properties - Considering the 1980-1990 time period, adjustments were made in the material properties to reflect improvements anticipated for these materials. In making the adjustments, no great revelations have been forecasted. Rather, it has been assumed that as a minimum, current inconsistencies in the material properties would diminish through refined processing. As a general guide, the 1980-1990 design values where obtained by assuming these values equal to the 1972-1973 mean property values. The resulting values for boron/polyimide and graphite/polyimide are compared to current values in Table 20-1. The data reported in Reference 3 provided the basis for the design values of both time periods.

No distinction has been made for room temperature and elevated temperature properties since, for the elevated temperatures of interest to this study, any variation in significant properties is within the accuracy of forecasted values.

Figures 20-2 and 20-3 presents comparative extensional modulus (E_{xx}) and shear modulus (G_{xy}) data for boron-epoxy, boron-polyimide, graphite-epoxy and graphite-polyimide. The data are room temperature properties for various percentage

TABLE 20-1. COMPOSITE DESIGN PROPERTIES FOR THE 1980-1990 TIME PERIOD

MATERIAL SYSTEM		BORON/PI		GRAPHITE/PI	
TIME PERIOD		1975	1985	1975	1985
PROPERTY					
E_1	10^6psi	32.2	32.2	22.8	22.8
E_2	10^6 psi	2.0	2.0	1.98	1.98
G_{12}	10^6 psi	0.80	0.8	0.71	0.71
ν_{12}		0.31	0.31	0.33	0.33
ν_{21}		0.019	0.019	0.029	0.029
α_1	10^{-6} in./in./F	2.8	2.8	-0.17	-0.17
α_2	10^{-6} in./in./F	15.0	15.0	9.45	9.45
ULTIMATE STRAINS					
ϵ_1(T)	10^{-6} in./in.	6,340	7,500	7,100	8,250
ϵ_2(T)	10^{-6} in./in.	2,560	3,000	4,400	4,950
ϵ_1(C)	10^{-6} in./in.	7,100	9,750	6,700	8,250
ϵ_2(C)	10^{-6} in./in.	4,700	9,750	8,250	8,250
γ_{12}	10^{-6} in./in.	15,000	15,000	15,000	15,000
ρ	lb/cu in.	0.072		0.056	
PLY THICKNESS (in.)		0.0055		0.007	

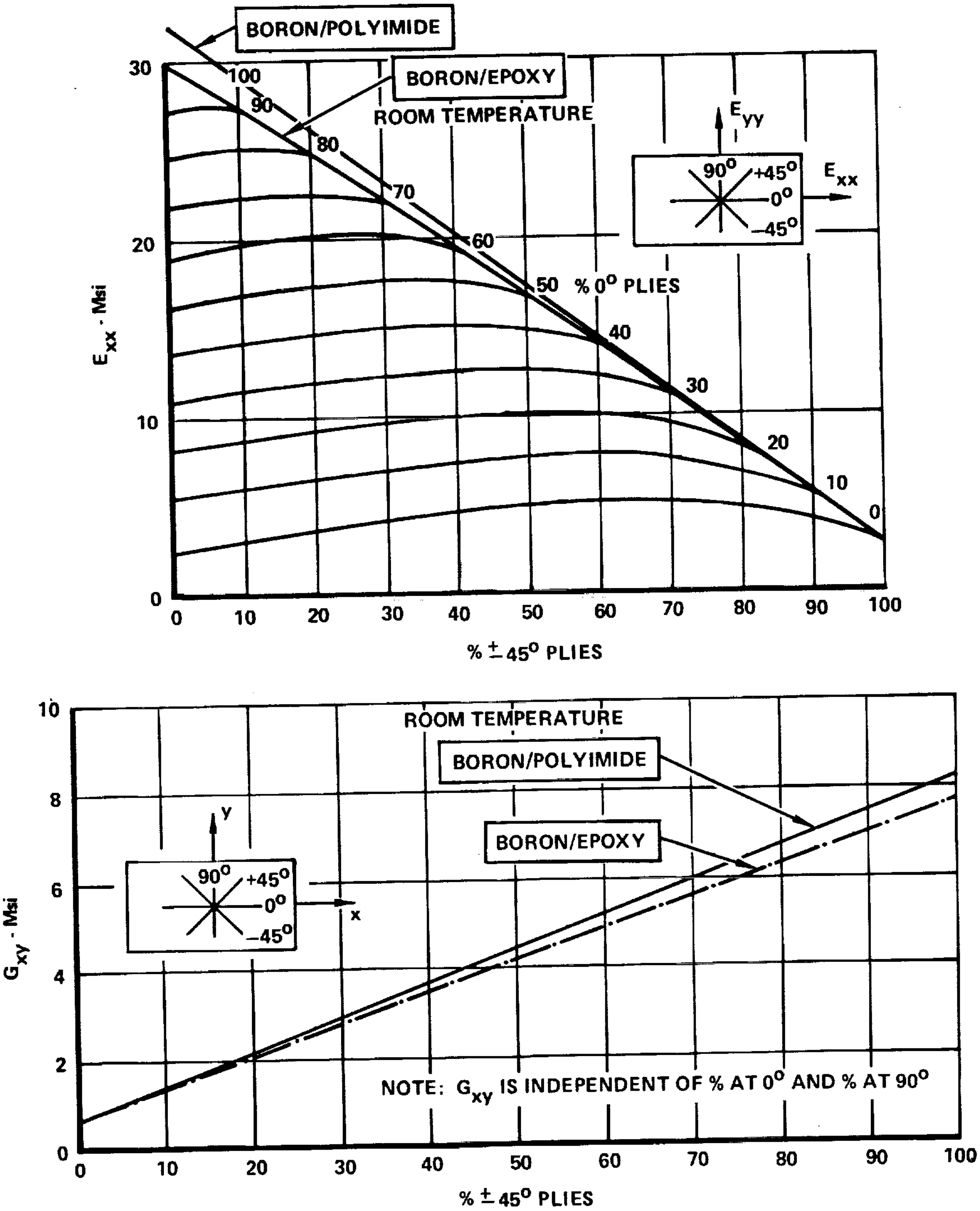


Figure 20-2. Extensional and Shear Modulus for Boron/Epoxy and Boron/ Polyimide

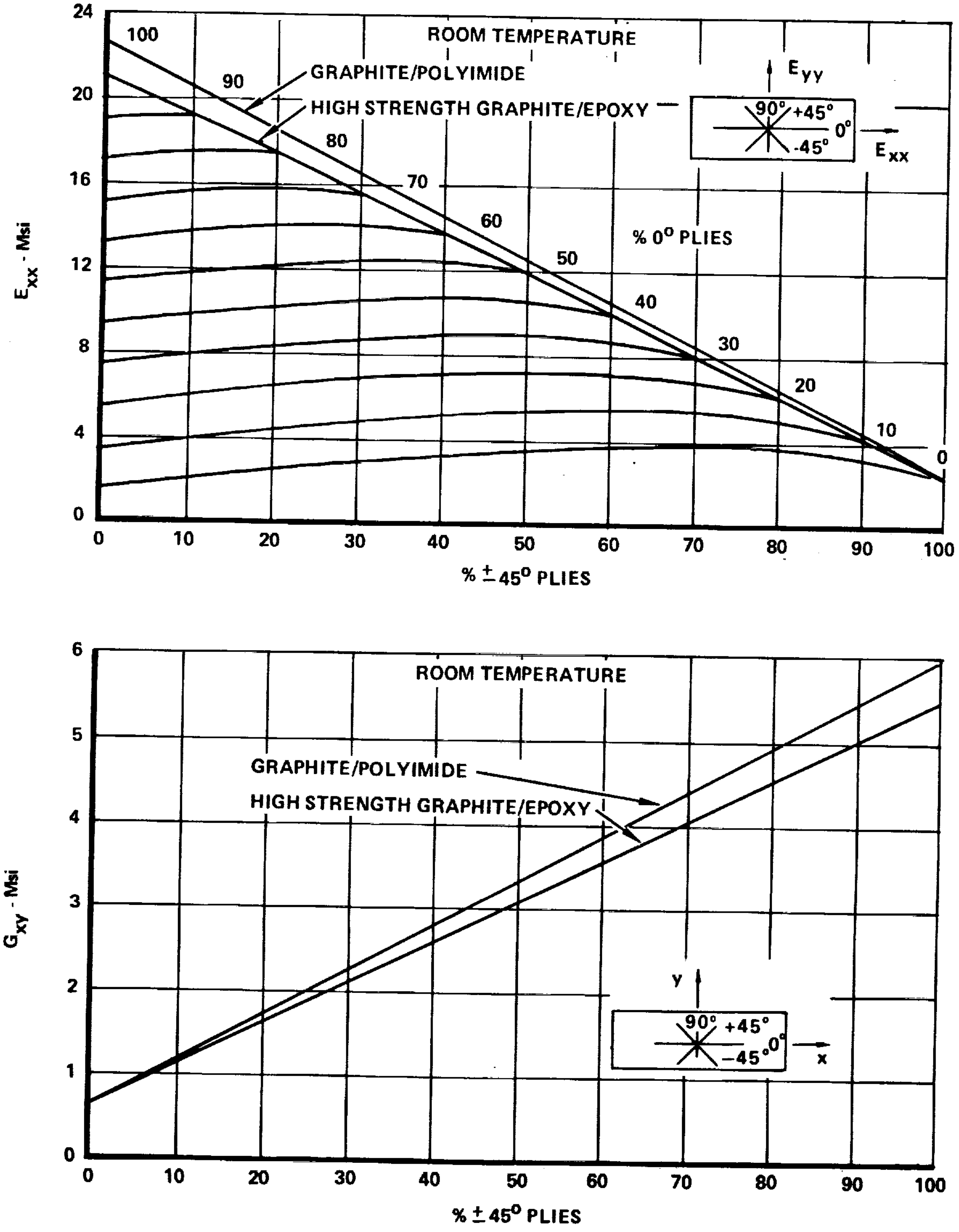


Figure 20-3. Extensional and Shear Modulus for Graphite/Epoxy and Graphite/Polyimide

of ±45-degree layup or combination of ±45-degree and 0-degree plies. Improvement in the polyimide material system properties are shown by the trends presented.

The thermal stress potential as measured by the product of the modulus of elasticity and coefficient of expansion is presented in Figure 20-4 versus percent of ±45-degree plies for the boron-polyimide and graphite-polyimide material systems. The potential for very small thermal stresses are noted by the near-zero $E\alpha_x$ for the graphite-polyimide system. For comparison purposes, the $E\alpha$ for titanium alloy 6Al-4V is also indicated. This figure does not reflect the thermal stresses that exist in an unrestrained cross-plied composite material due to change in temperature and due to different coefficient of expansion, which does not occur in titanium.

Figure 20-5 presents specific stiffness data for both the B/PI and Gr/PI as compared to the reference titanium system. Four-fold improvement in properties are indicated for both composite material system with the most significant improvement offered by the boron-polyimide material system. For buckling critical design, similar trends are indicated because of the improved modulus to density ratio.

Protective System - It is recognized that composite materials require provisions for protection beyond that of all-metal counterparts. In particular it is desirable to protect against degradation by aggressive environments such as electrical hazards, erosion, impact, and weathering. The basic external protective system arrangement assumed for this study is illustrated in Figure 20-6.

For lightning and static electricity problems the optimum protection, weight, and producibility is offered by the aluminum wire mesh. Two wire-mesh configurations have been provided. All exterior surfaces are to be covered with 200 x 200 mesh except the edge sections where 120 x 120 mesh is to be applied. The 120 x 120 mesh was selected for the leading edge to improve heat dissipation for this area. Additional composite protection is provided by an electrical insulating barrier consisting of one ply of 120 glass laminated between the laminate and the aluminum wire mesh. The 120 glass barrier ply and the aluminum wire mesh are cocured with the resin from the 120 glass, bonding the mesh to the composite. The wire mesh is connected to metallic substructure to provide a path for electrical discharge.

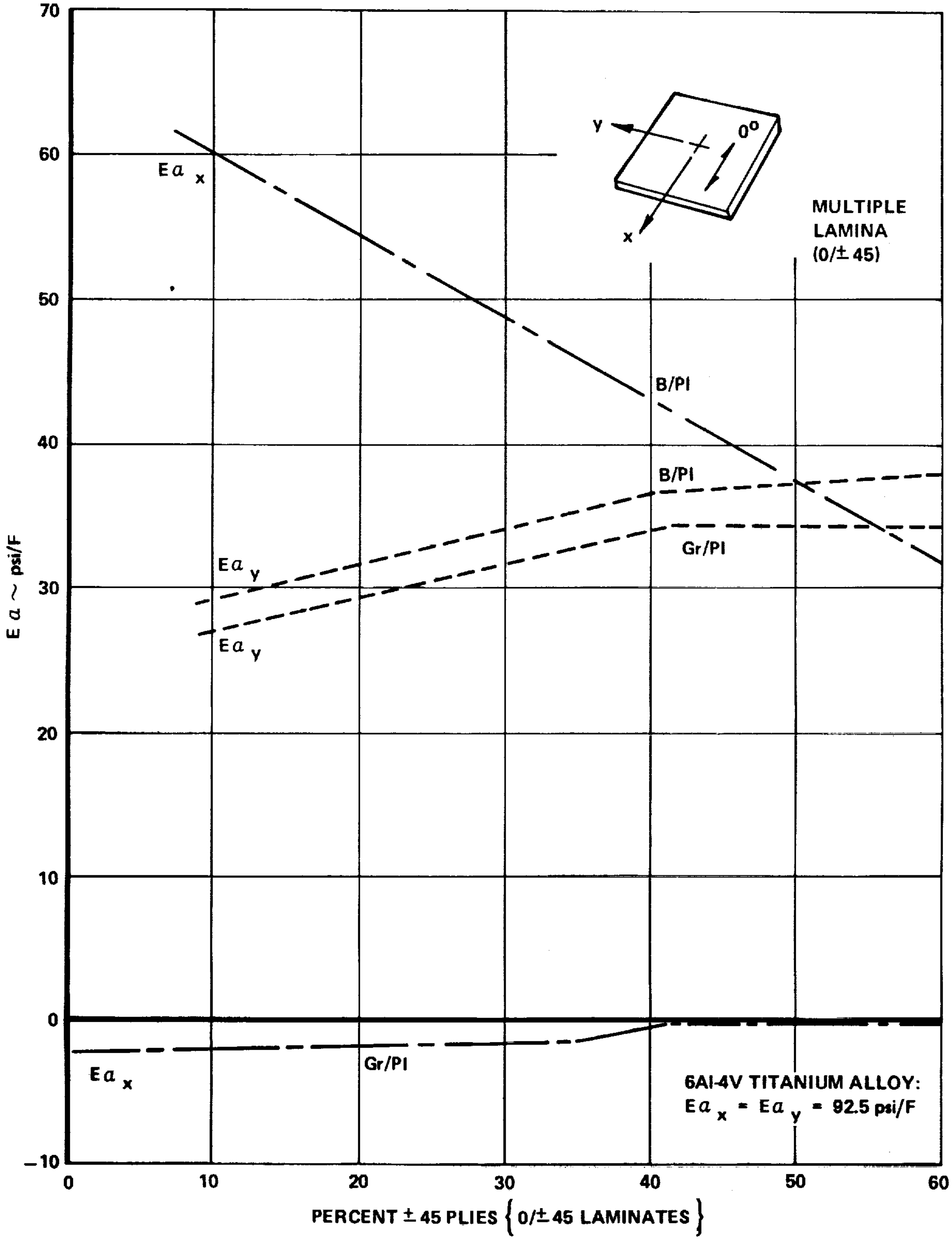


Figure 20-4. Comparative Thermal Stress - Titanium vs Composite Materials

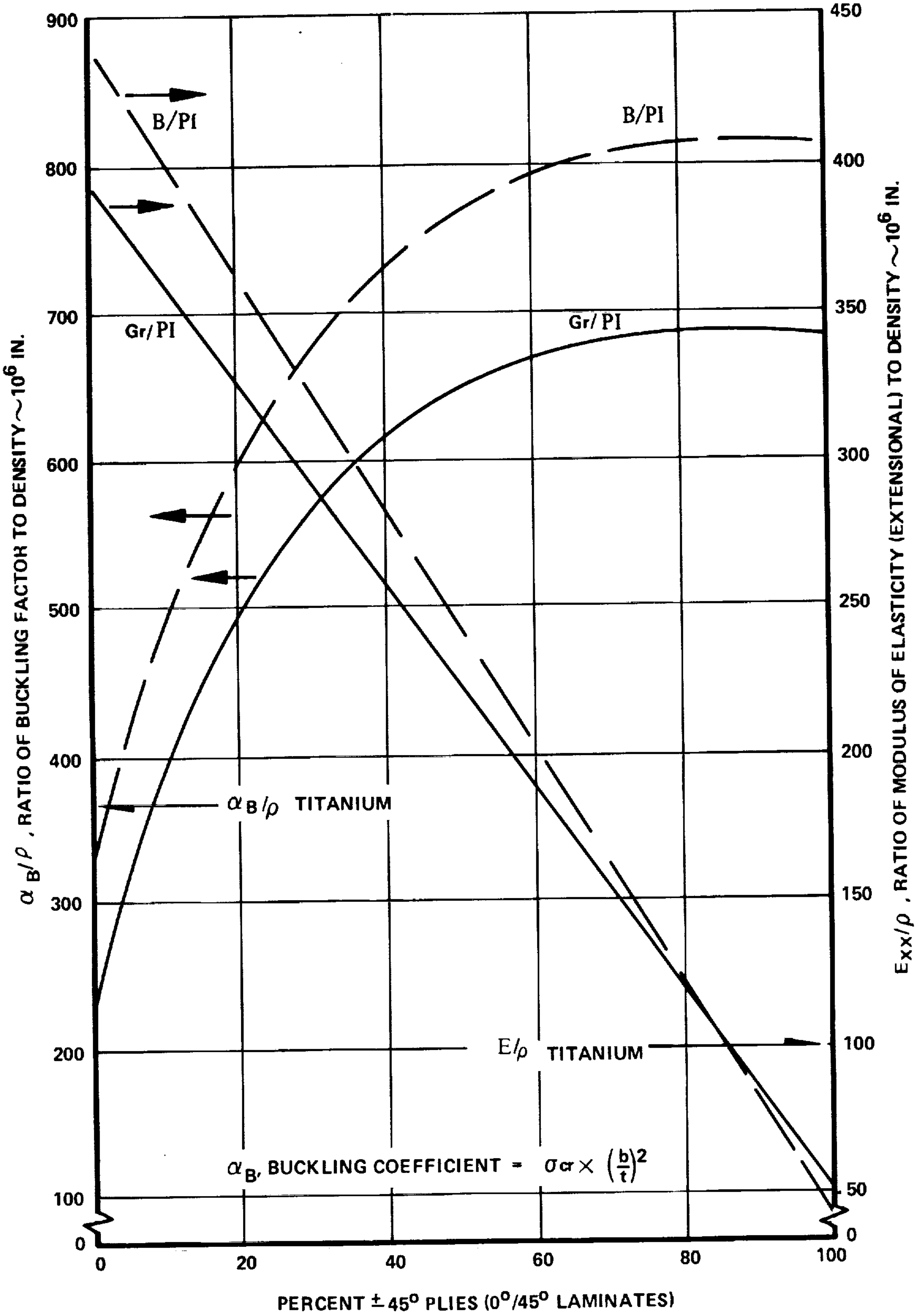


Figure 20-5. Comparative Buckling Efficiency and Specific Stiffness for Boron/Polyimide and Graphite/Polyimide

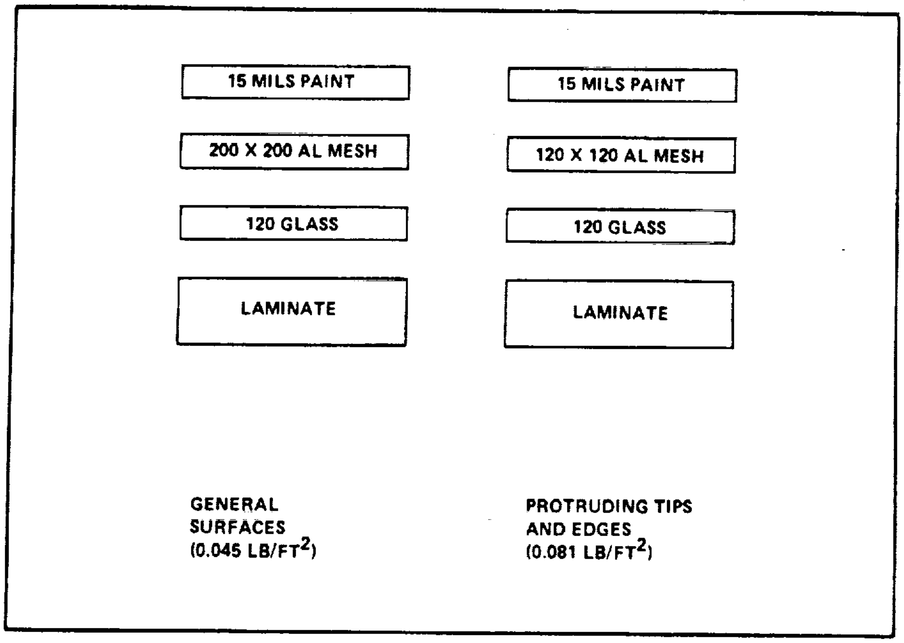

15 MILS PAINT
200 X 200 AL MESH
120 GLASS
LAMINATE
15 MILS PAINT
120 X 120 AL MESH
120 GLASS
LAMINATE
GENERAL SURFACES (0.045 LB/FT²)
PROTRUDING TIPS AND EDGES (0.081 LB/FT²)

Figure 20-6. Surface Protection System for Composite Laminates

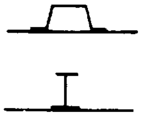
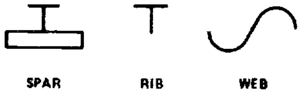
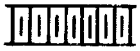
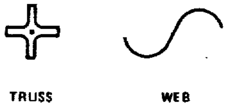
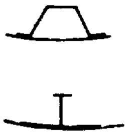
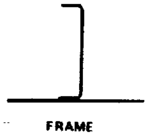
STRUCTURAL ARRANGEMENTS	BASIC CONCEPTS	
	SURFACE PANEL	SUBSTRUCTURE
WING CHORDWISE STIFFENED		 SPAR RIB WEB
WING MONOCOQUE		 TRUSS WEB
FUSELAGE SKIN/STRINGER AND FRAME		 FRAME

Figure 20-7. Composite Wing and Fuselage Design Concepts

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Other protective measures include coating all surfaces with a polyurethane system (more desirable, higher service temperature systems are anticipated by 1985) sealing all cut edges, and wet installation of fasteners. The amount of protection required varies with the component location, with the greatest care required for leading-edge structure. Maximum protection, therefore, is assumed for these areas.

Design Concepts

The primary load carrying composite concepts for the far-term supersonic cruise aircraft design explored both biaxially and uniaxially stiffened surface panels. The concepts examined were variations of those evaluated in Task I, thus, the internal loads calculated for the Analytical Design Studies were directly applicable to this technology assessment study.

Figure 20-7 presents the basic design concepts evaluated for both wing and fuselage designs. All wing surface panel concepts are smooth skin designs and exploit the low coefficient of expansion characteristics, especially inherent in the graphite-polyimide system. For the fuselage, the more conventional skin-stringer and frame designs are evaluated. Further amplification of design variations are discussed in Table 20-2 and pictorially displayed in Figure 20-8.

Point Design Regions

Representative structure was specified at selective wing and fuselage regions. Analysis of the selected regions are performed to establish unit weights which are employed to establish total airplane weights.

Wing Point Design Regions - The location of wing point design regions are shown in Figure 20-9 and include the 3 regions which are displayed on the wing planform of the structural model. Point design regions are identified by the corresponding NASTRAN panel element numbers. Representative structure is specified at each of these locations and include a definition of the upper and lower surface panels, typical rib and spar structure, and the associated non-optimum factors. These regions were selected as representative of wing critical design regions. A description of these regions is as follows:

TABLE 20-2. MERITS OF POTENTIAL ALL-COMPOSITE CHORDWISE STIFFENED DESIGN CONCEPTS

CONCEPT A – CORRUGATED HAT STIFFENER

PROVIDES GOOD COMPRESSION EFFICIENCY, ESPECIALLY WITH EXTRA UNIDIRECTIONAL MATERIAL APPLIED IN HAT CROWN ELEMENT. CLOSED STIFFENER PROVIDES EXCELLENT TORSIONAL RIGIDITY AND LATERAL STABILITY PROPERTIES.

CONCEPT B – POINTED HAT STIFFENER

ALLOWS EFFICIENT PLACEMENT OF COMPRESSIVE MATERIAL, BUT REPRESENTS A REDUCTION IN LATERAL STABILITY FOR A CLOSED SECTION STIFFENER. SPAR CAP INTERFACE REPRESENTS WEIGHT PENALTY.

CONCEPT C – ROUND HAT STIFFENER

ALTHOUGH A ROUNDED SECTION REPRESENTS ADDED BUCKLING STRENGTH, THE INCREASE IS THOUGHT TO BE LESS FOR A LAMINATED COMPOSITE ELEMENT RELATIVE TO A CIRCULAR METAL STIFFENER. MANUFACTURING DIFFICULTIES AND SPAR CAP INTERFACE REPRESENTS ADDITIONAL PROBLEM AREAS.

CONCEPT D – HONEYCOMB STABILIZED HAT STIFFENER

THE RIGIDITY OF THE HONEYCOMB PRESENTS TOLERANCE PROBLEMS IN BOND AND CURE OF THE PANEL. THE CORE/SKIN INTERFACE UNDER THE HAT REPRESENTS A "HARD SPOT" RESTRAINING SMOOTH EXPANSION AND CONTRACTION, ESPECIALLY FOR BORON-POLYIMIDE MATERIAL.

CONCEPT E – TEE STIFFENER

FOR LIGHT WEIGHT COMPOSITE PANELS, TEE STIFFENERS OF MULTIDIRECTIONAL LAYERS REPRESENT A VERY WEAK TORSIONAL STABILITY CAPABILITY. TOTAL SKIN THICKNESS INCREASED RELATIVE TO A CLOSED STIFFENER LAYOUT.

CONCEPT F – BULB STIFFENER

ALTHOUGH COMPRESSION EFFICIENCY IS ENHANCED BY THE JUDICIOUS PLACEMENT OF A BUNDLE OF UNIDIRECTIONAL FIBERS, HERE AGAIN TORSIONAL RIDIGITY AND LATERAL STABILITY SUFFERS. ALSO, SPAR CAP INTERFACE WEIGHT PENALTIES WILL RESULT.

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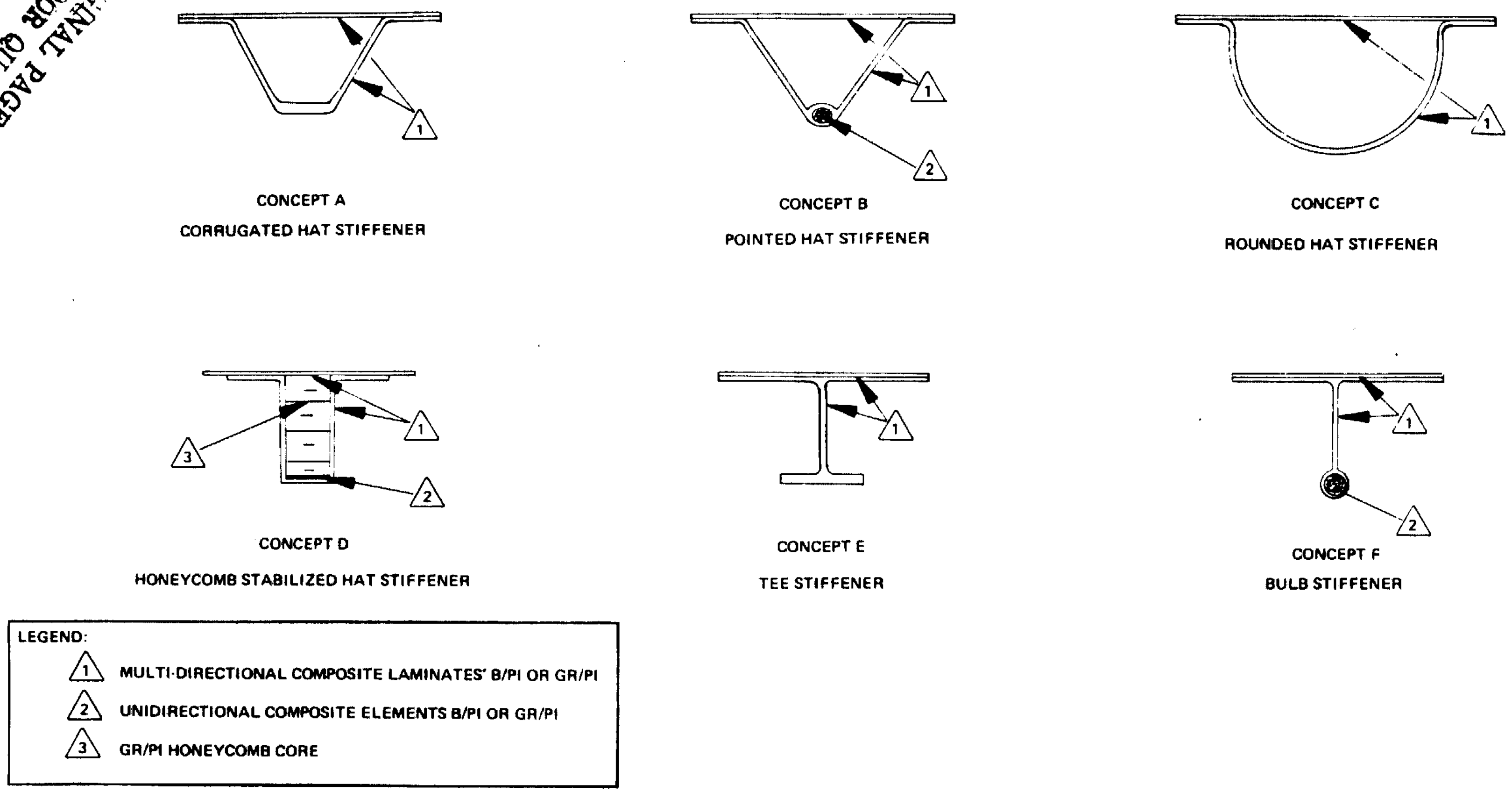

CONCEPT A
CORRUGATED HAT STIFFENER

CONCEPT B
POINTED HAT STIFFENER

CONCEPT C
ROUNDED HAT STIFFENER

CONCEPT D
HONEYCOMB STABILIZED HAT STIFFENER

CONCEPT E
TEE STIFFENER

CONCEPT F
BULB STIFFENER

LEGEND:
1 MULTI-DIRECTIONAL COMPOSITE LAMINATES' B/PI OR GR/PI
2 UNIDIRECTIONAL COMPOSITE ELEMENTS B/PI OR GR/PI
3 GR/PI HONEYCOMB CORE
1
2
3

Figure 20-8. Chordwise Stiffened Wing Design - All Composite Concepts

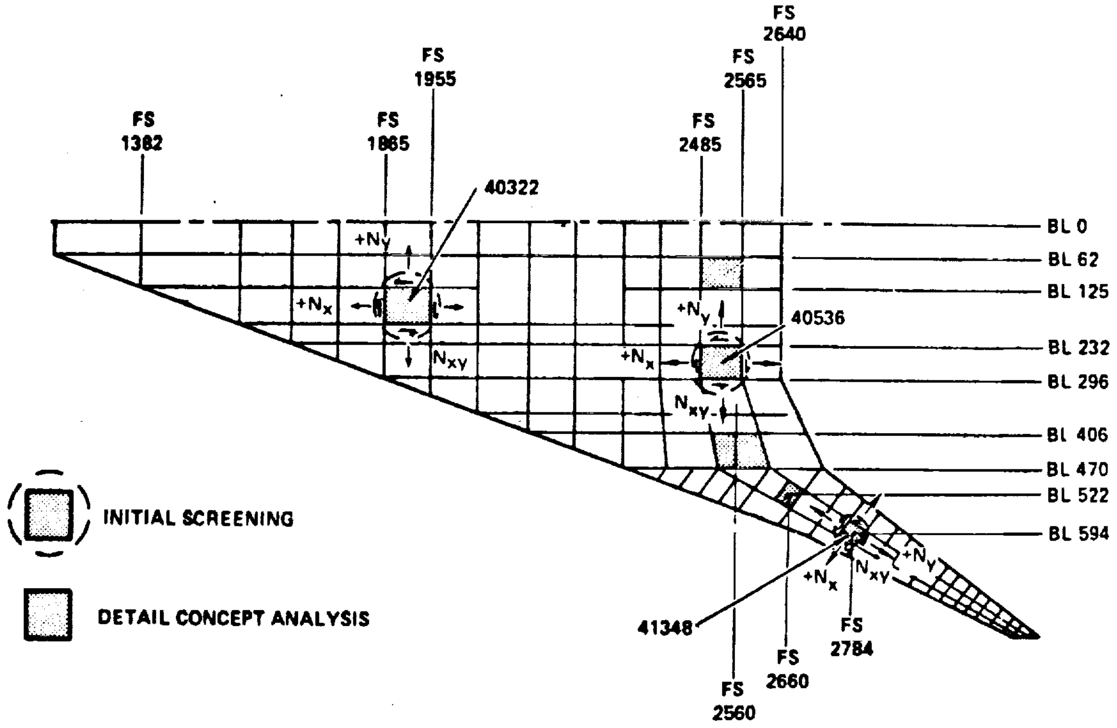
FS 2640
FS 2565
FS 1955
FS 2485
FS 1382
FS 1865
40322
+Ny
+Nx
Nxy
+Ny
40536
+Nx
Nxy
+Ny
+Nx
Nxy
BL 0
BL 62
BL 125
BL 232
BL 296
BL 406
BL 470
BL 522
BL 594
INITIAL SCREENING
DETAIL CONCEPT ANALYSIS
41348
FS 2784
FS 2660
FS 2560

Figure 20-9. Definition of Wing Point Design Regions

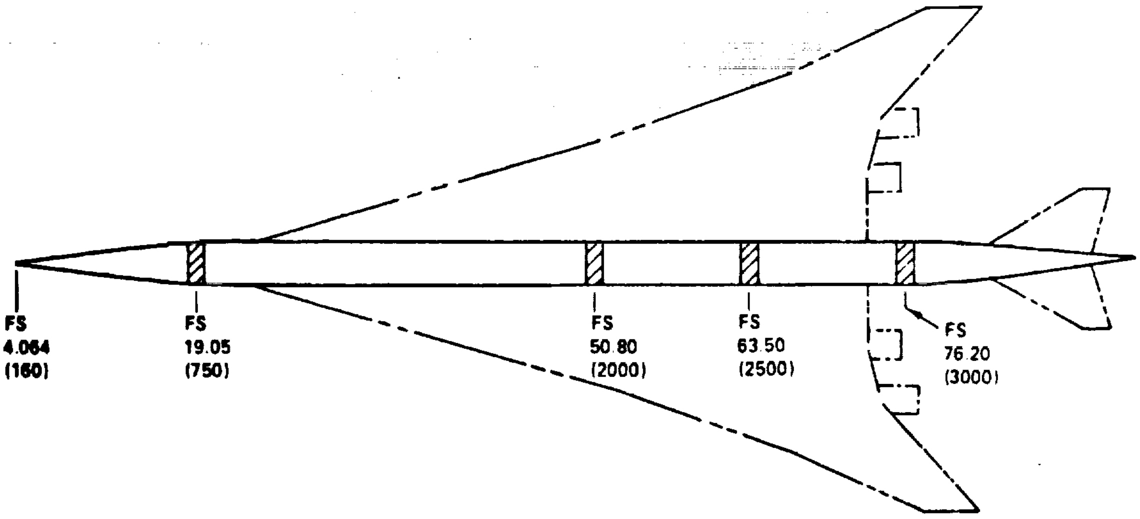
FS 4.064 (160)
FS 19.05 (750)
FS 50.80 (2000)
FS 63.50 (2500)
FS 76.20 (3000)

Figure 20-10. Definition of Fuselage Point Design Regions

- Forward wing box - Point design region 40322 is located forward of the main landing gear in a fuel tank region. This area is characterized as basically transmitting pressure loads with low load intensities with respect to wing bending loads.

- Aft box region - Point design region 40536, is located in the wing aft box in fuel tank region. In general, this area represents regions of high spanwise load intensities and variable chordwise load intensities due to wing bending.

- Wing tip region - Dry bay region 41348 is located approximately mid-span of the wing tip. High load intensities are indicative of the aeroelastic effect on this flexible region.

Fuselage Point Design Regions - Four point design regions were selected as representative of the actual fuselage design. These regions are shown in Figure 20-10 and are located at fuselage stations 750, 2000, 2500, and 3000. These regions were selected as typical of the critical design regions on the fuselage and, in general, classified as follows:

- Fuselage Forebody FS 750 - Generally characterized as fatigue design structure with low load intensities due to fuselage bending.

- Fuselage Centerbody (FS 2000 and 2500) - Wing/fuselage regions subjected to maximum body bending and wing spanwise loads.

- Fuselage Aftbody (FS 3000) - High body bending and torsion loads with regions subjected to a high acoustic environment.

Fuselage point design regions located at FS 2000 and FS 2500 are coincidental with the wing forward box and aft box point design regions.

Wing Design Loads

The Task I internal loads and surface pressures were scanned to identify the potentially critical conditions for wing and fuselage design. Table 20-3 presents the flight parameters associated with the selected design condition.

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TABLE 20-3. CRITICAL WING LOADING CONDITIONS

LOAD CONDITION	WEIGHT (LB)	V_e (KEAS)	MACH NO.	ALTITUDE (10^3 FT.)	n_z
13	700,00	325	.90	30	2.5
20[1]	660,000	460	2.70	61.5	2.5
22[2]	550,000	433.6	2.70	64	2.5
31	690,000	265	1.25	52.4	2.5

[1] Start-of-Cruises [2] Mid-Cruise

The surface load intensities and corresponding pressures are shown in Table 20-4 and 20-5 for the chordwise stiffened and monocoque designs, respectively. Load reduction based on reduced airframe weight potential of advanced composites application was not included.

Fuselage Design Loads

The fuselage design loads used for the composites design evaluation are in accordance with the data of Figure 20-11 and 20-12. The shears and bending moments were used to size panels and frames at the specified point design regions. In addition, the stiffness of the composite shell was maintained at least equivalent to the titanium shell design.

Results-Wing

Chordwise Stiffened Design Concept - Screening of the potential all-composite design concepts of Table 20-2 and Figure 20-8 were performed both on a qualitative and quantitative basis. The results of this assessment identified the Corrugated Hat Stiffener (Concept A) and the Tee Stiffener (Concept E) as the leading candidate for the all-composite design. The former provides good compression efficiency and the closed section offers excellent torsional rigidity. For the more lightly loaded, pressure critical, forward wing box structure the tee-stiffener concept was also evaluated. Although a torsionally-weak section, proper detail design often provides adequate structural integrity with minimum weight.

TABLE 20-4. DESIGN LOADS FOR CHORDWISE STIFFENED ARRANGEMENT - ULTIMATE

LOAD CONDITION			POINT DESIGN REGION					
			40322		40536		41348	
			UPPER	LOWER*	UPPER	LOWER*	UPPER	LOWER*
13	N_x	(lb/in.)	-531		271		-633	
	N_y	(lb/in.)	-1,115		-9,777		-8,007	
	N_{xy}	(lb/in.)	-236		-778		1,896	
	p	(psi)	-8.14	-9.26	-6.98	-8.40	-3.86	-0.30
20	N_x	(lb/in.)	-131		-527		-265	
	N_y	(lb/in.)	-866		-8,740		-7,415	
	N_{xy}	(lb/in.)	-97		-1,187		855	
	p	(psi)	-7.67	-9.47	-7.47	-7.47	-1.29	1.04
22	N_x	(lb/in.)	396		-152		-225	
	N_y	(lb/in.)	-775		-7,355		-6,670	
	N_{xy}	(lb/in.)	-49		-1,016		745	
	p	(psi)	-6.65	-7.16	-5.80	-5.00	-1.06	0.86
31	N_x	(lb/in.)	488		-1,306		-1,402	
	N_y	(lb/in.)	-1,063		-14,379		-12,979	
	N_{xy}	(lb/in.)	-120		-2,364		3,144	
	p	(psi)	-8.33	-8.94	-6.94	-8.29	-5.07	0.96

*REVERSE SIGN OF UPPER SURFACE EXCEPT AS NOTED.

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TABLE 20-5. DESIGN LOADS FOR MONOCOQUE ARRANGEMENT - ULTIMATE

LOAD CONDITION			POINT DESIGN REGION					
			40322		40536		41348	
			UPPER	LOWER*	UPPER	LOWER*	UPPER	LOWER*
13	N_x	(lb/in.)	-452		-1,218		-804	
	N_y	(lb/in.)	-621		-7,953		-5,908	
	N_{xy}	(lb/in.)	-210		-1,903		2,657	
	p	(psi)	-8.14	-9.26	-6.98	-8.40	-3.86	-0.3
20	N_x	(lb/in.)	-213		-1,945		-342	
	N_y	(lb/in.)	-463		-7,257		-6,037	
	N_{xy}	(lb/in.)	-135		-2,588		1,499	
	p	(psi)	-7.67	-9.47	-7.47	-7.47	-1.29	1.04
22	N_x	(lb/in.)	69		-1,381		-290	
	N_y	(lb/in.)	-378		-6,081		-5,381	
	N_{xy}	(lb/in.)	-109		-2,196		1,305	
	p	(psi)	-6.65	-7.16	-5.8	-5.0	-1.06	0.86
31	N_x	(lb/in.)	51		-3,272		-1,578	
	N_y	(lb/in.)	-529		-11,787		-9,586	
	N_{xy}	(lb/in.)	-191		-4,795		4,225	
	p	(psi)	-8.33	-8.94	-6.94	-8.29	-5.07	0.96

*REVERSE SIGN OF UPPER SURFACE EXCEPT AS NOTED.

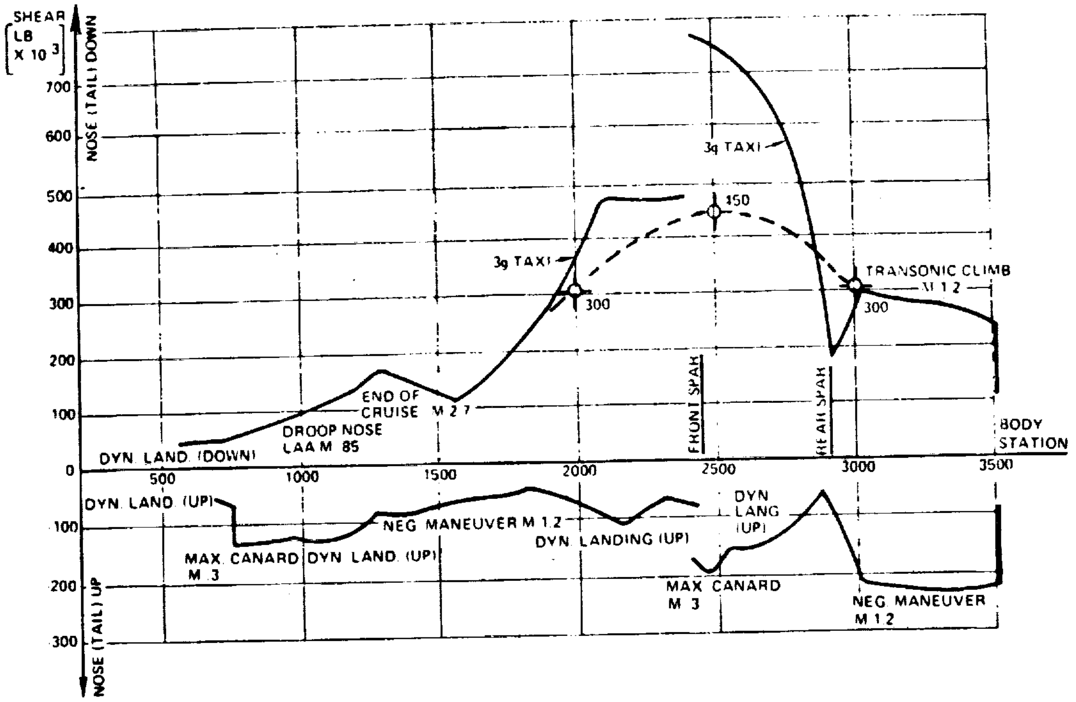

Figure 20-11. Fuselage Shear Diagram - Task I

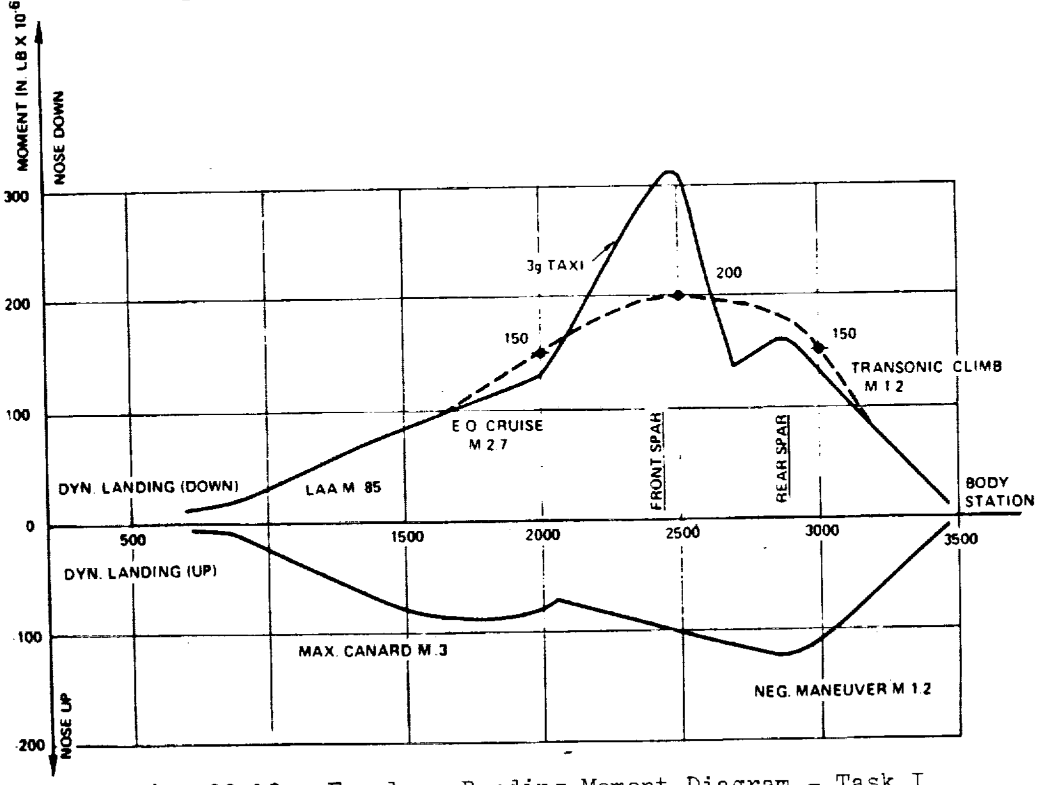

Figure 20-12. Fuselage Bending Moment Diagram - Task I

Figure 20-13 presents a layout of a typical section of the wing for the chordwise stiffened design. Representative truss and bulkhead spars are indicated. Both a honeycomb and corrugated bulkhead-type spar/rib web are conceptually shown. A summary of the composite wing surface panel designs are shown in Table 20-6. The geometry and layup for the hat-stiffener for point design region 40322 and 40536 are presented. For both the boron-polyimide and the graphite-polyimide material system, the same geometry is used for a given region. The ply thickness and density of the graphite and boron material varies thus resulting in slight variations in equivalent panel thickness and panel unit weight. The detail geometry for the tee-stiffener design indicates the reduced efficiency and thus higher unit weight by approximately 40-percent.

The substructure sizing results are displayed in Table 20-7 for the graphite-polyimide material system at point design region 40322. The details of representative substructure components (i.e. spar caps, rib caps, spar and rib webs) are tabulated. The geometry and layup orientation employed in the design are also shown. Proportionate unit weights were developed for the boron-polyimide substructure at the various point design regions.

Monocoque Design Concept - For the monocoque design only the honeycomb sandwich was evaluated. The design of the sandwich panel was based on laminated face skins of boron-polyimide and graphite-polyimide composites with a titanium alloy core. Condition 13 was critical for point design 40322 and condition 31 critical for both the aft box and tip box structure regions.

Figure 20-14 presents a layout of a typical wing section employing the honeycomb surface panel design. Although a total honeycomb system is pictorially displayed, the substructure weights are for representative truss spar/rib and corrugated spar/web designs. The details for the wing surface panels are shown in Table 20-8 for the B/PI and Gr/PI designs. The resulting panel unit weights include a protective system weight of .045-lb per square foot.

For the truss spar and rib design, a cruciform configuration was adopted in lieu of the tubular section used for the metallic design. The composite cruciform member was sized to yield a greater cross-section "EI" at the same column length as its titanium counterpart. Comparative data is shown in Table 20-9 for the titanium truss tubes and the composite section. A weight saving of approximately

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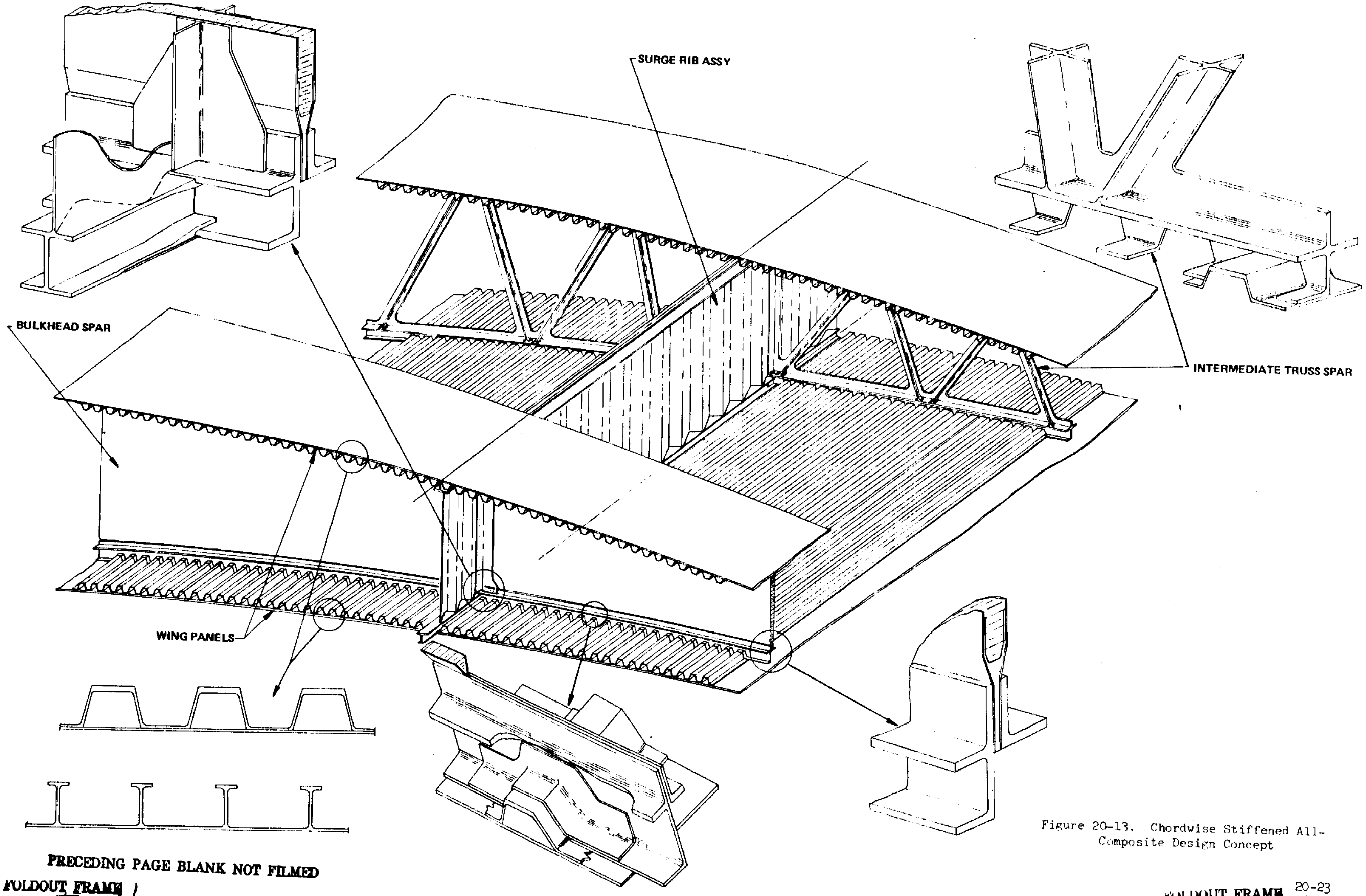
SURGE RIB ASSY
BULKHEAD SPAR
INTERMEDIATE TRUSS SPAR
WING PANELS

Figure 20-13. Chordwise Stiffened All-Composite Design Concept

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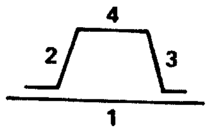
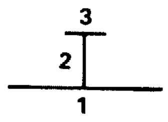
ELEMENT CODES							
REGION		40322		40536		40322	
MATERIAL		B/PI	Gr/PI	B/PI	Gr/PI	B/PI	Gr/PI
ELEMENT							
1	LENGTH (in.)	3.0		2.5		2.5	
	ORIENTATION	33% 0° – 67% ±45°		20% 0° – 80% ±45°		20% 0° – 80% ±45°	
	PLIES	6		10		10	
2	LENGTH (in.)	0.5		0.5		1.2	
	ORIENTATION	33% 0° – 67% ±45°		50% 0° – 50% ±45°		20% 0° – 80% ±45°	
	PLIES	6		8		10	
3	LENGTH (in.)	1.0		1.0		1.2	
	ORIENTATION	33% 0° – 67% ±45°		50% 0° – 50% ±45°		55% 0° – 45% ±45°	
	PLIES	6		8		18	
4	LENGTH (in.)	1.0		1.0			
	ORIENTATION	71% 0° – 29% ±45°		65% 0° – 35% ±45°			
	PLIES	14		23			
SPACING (in.)		3.0		2.5		2.5	
PANEL $\bar{t}$ (in.)		0.092	0.117	0.159	0.202	0.129	0.164
WEIGHT$^{(A)}$, $\rho\bar{t}$ (lb/ft^2)		0.998	0.988	1.694	1.674	1.384	1.370

(A) INCLUDES PROTECTIVE FINISH.

TABLE 20-6. SUMMARY OF COMPOSITE CHORDWISE STIFFENED WING SURFACE PANELS

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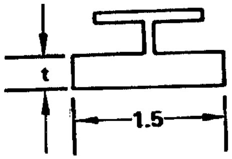
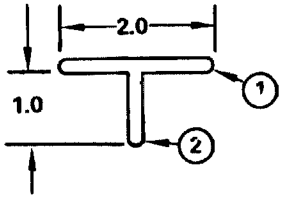
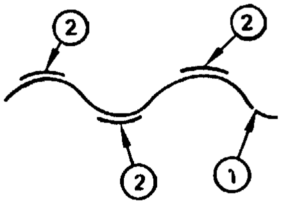
TABLE 20-7. GRAPHITE-POLYIMIDE SUBSTRUCTURE - POINT DESIGN 40322

SUBSTRUCTURE COMPONENT	GEOMETRY	ORIENTATION	SURFACE OR ELEMENT	t (in.)	PLIES (n)	AREA (in^2)	ρA (lb/in.)	ω (lb/ft^2)
SPAR CAPS		$0_{32}/\pm45_{32}$	UPPER	0.32	64	0.48	0.0264	0.127
		$0_{52}/\pm45_{52}$	LOWER	0.52	104	0.78	0.0429	0.207
			Σ	0.84	168	1.26	0.0693	0.334
RIB CAPS		$0_{12}/\pm45_{8}$	1	0.140	20	0.280	0.0157	0.038
		$0_{4}/\pm45_{8}$	2	0.084	12	0.084	0.0047	0.011
			Σ X 2	0.448	64	0.728	0.0408	0.098
SPAR AND RIB WEBS		$0_{1}/\pm45_{4}$	1	0.035	5	0.130	0.0073	$0.411^{(A)}$
		0_{6}	2	0.042	6	0.042	0.0024	$0.135^{(A)}$
			Σ	0.077	11	0.172	0.0097	$0.546^{(A)}$

(A) UNIT WEIGHT OF WEB NOT COMPENSATED FOR DEPTH OF WEB OR SPACING. MULTIPLY BY 0.98 FOR UNIT SPAR WEIGHT; BY 1.06 FOR UNIT RIB WEIGHT.

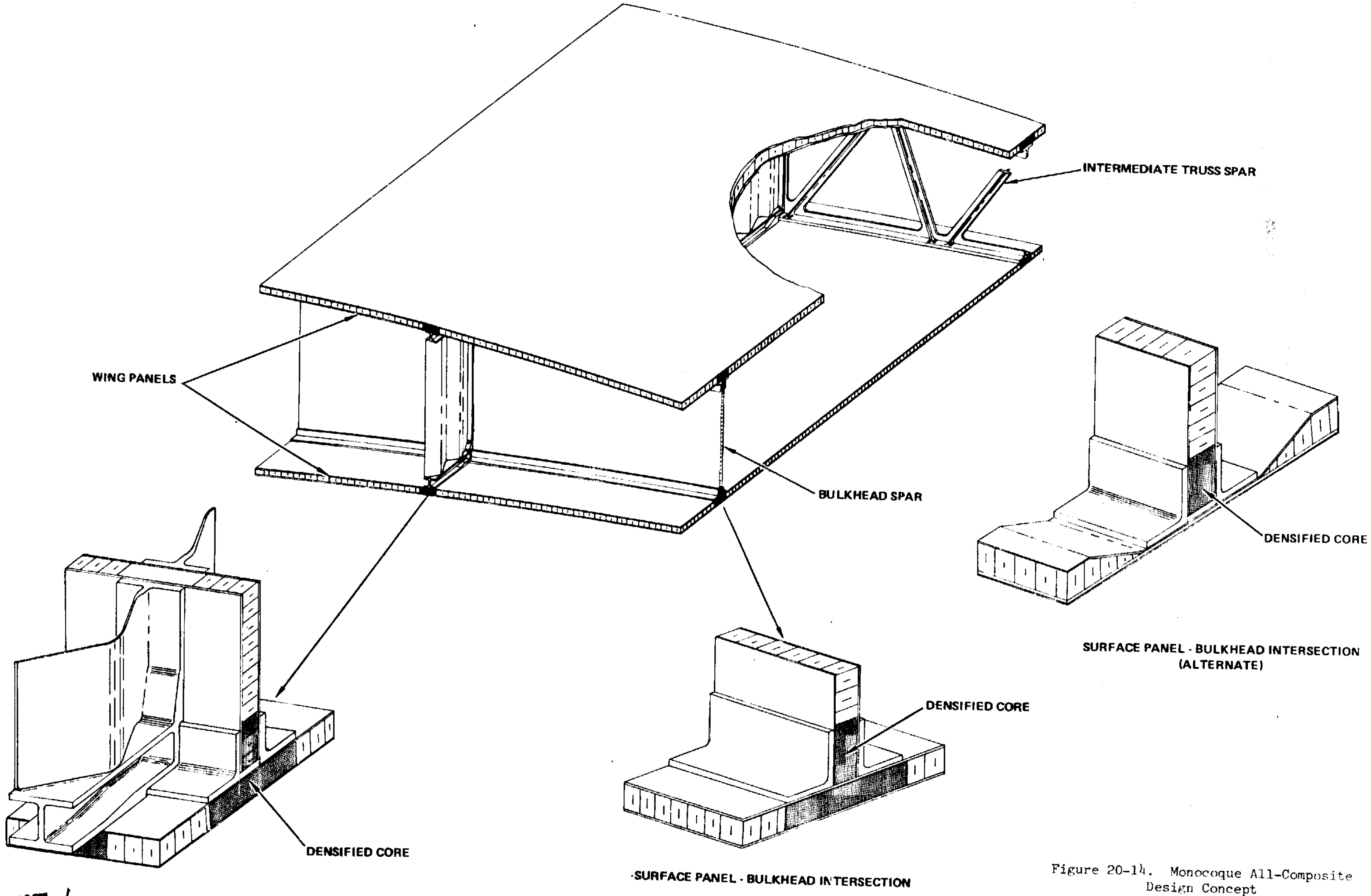

INTERMEDIATE TRUSS SPAR
WING PANELS
BULKHEAD SPAR
DENSIFIED CORE
SURFACE PANEL - BULKHEAD INTERSECTION (ALTERNATE)
DENSIFIED CORE
DENSIFIED CORE
SURFACE PANEL - BULKHEAD INTERSECTION

Figure 20-14. Monocoque All-Composite Design Concept

FOLDOUT FRAME 1

FOLDOUT FRAME 2

TABLE 20-8. SUMMARY OF COMPOSITE MONOCOQUE WING SURFACE PANELS

POINT DESIGN REGION		40322				40536				41348			
FACE MATERIAL		B/PI		GR/PI		B/PI		GR/PI		B/PI		GR/PI	
NO. PLIES/FACE		6	6	6	6	17	17	17	17	15	15	15	15
SURFACE		UPPER	LOWER	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER
ORIENTATION		$0_2/\pm45_2/90_2$				$0_3/\pm45_8/90_6$				$0_2/\pm45_8/90_5$			
SKIN	lb/ft^2	0.729	0.771	0.722	0.764	1.993	2.595	1.964	2.556	1.745	1.894	1.738	1.886
CORE	lb/ft^2	0.315	0.186	0.315	0.186	0.262	0.028	0.262	0.028	0.380	0.099	0.380	0.099
Σ PANEL (A)	lb/ft^2	1.044	0.957	1.037	0.950	2.255	2.623	2.226	2.584	2.125	1.993	2.118	1.985

(A) INCLUDES 0.045 lb/ft^2 PROTECTIVE FINISH WEIGHT

(B) SPAR SPACING = 30 INCHES
RIB SPACING = 60 INCHES

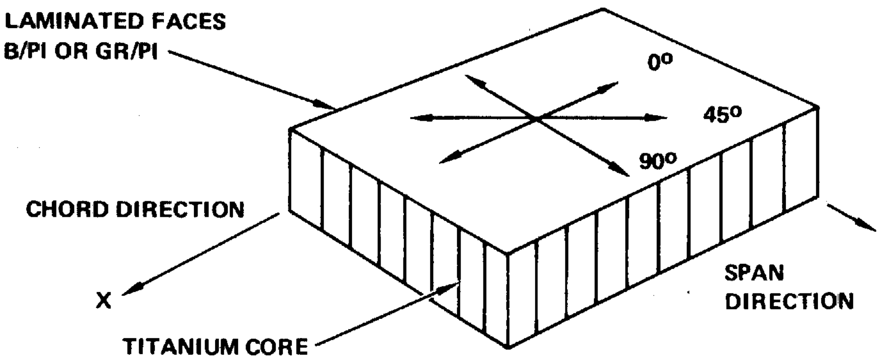
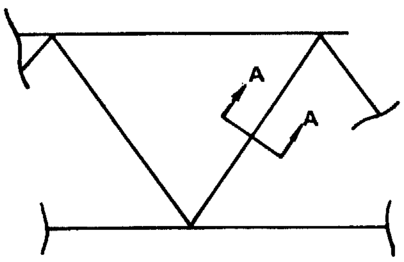
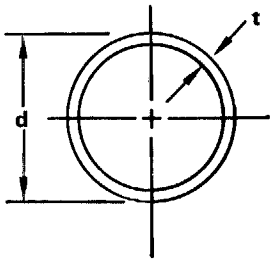

LAMINATED FACES
B/PI OR GR/PI
0°
45°
90°
CHORD DIRECTION
X
TITANIUM CORE
SPAN DIRECTION

TABLE 20-9. COMPOSITE TRUSS SPAR AND RIB - MONOCOQUE DESIGN

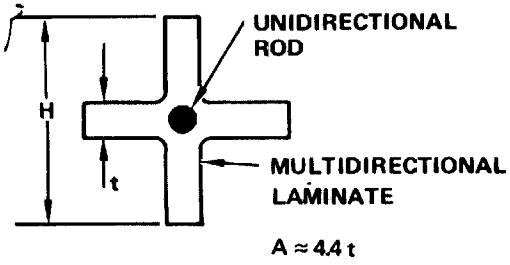
POINT DESIGN REGION	SPAC	TITANIUM TRUSS TUBES				COMPOSITE CRUCIFORM TRUSS						
		A	ρA	EI X 10^6	w	H	t	A	ρA	EI X 10^6	wB/PI	wGr/PI
	(in.)	(in^2)	(lb/in.)	(lb/in^2)	(lb/ft^2)	(in.)	(in.)	(in^2)	(lb/in.)	(lb/in^2)	(lb/ft^2)	(lb/ft^2)
40322												
SPAR TRUSS	30	.093	.0149	0.40	0.284	2.0	0.044	.194	.0139	0.70	0.265	0.267
RIB TRUSS	30	.329	.0526	1.92	0.387	2.0	0.154	.678	.0488	2.50	0.359	0.362
40536												
SPAR TRUSS	20	.271	.0434	0.96	0.586	2.0	0.121	.532	.0383	1.94	0.517	0.523
RIB TRUSS	20	.709	.1134	5.02	0.406	2.0	0.308	1.256	.0976	5.06	0.349	0.354



WING TRUSS



SECT. AA


UNIDIRECTIONAL
ROD
H
t
MULTIDIRECTIONAL
LAMINATE
A ≈ 4.4 t

SECT. AA

6-percent is realized over the titanium alloy design. Note the unidirectional rod used in the center of the cruciform for achieving a high EA product.

The corrugated composite spar and rib webs were sized to provide a greater $\bar{Et}$ or $\bar{Gt}$ than the titanium alloy design. This approach offered reasonable pressure (bending) and shear capability, respectively. The overall corrugation dimensions were retained for the composite webs. Table 20-10 presents the geometry data developed for the circular-arc corrugated spar/rib webs for the reference titanium alloy. The appropriate geometry for the 3 point design regions are shown. The results of the composite spar web design at point design region 40322 is shown in Table 20-11. The results indicate a weight saving of approximately 25-percent when composite designs are employed. Since the cross section properties for the spar and rib webs in regions 40322, 40536 and 41348 are of the same magnitude, the region 40322 spar web weight saving factor was used to calculate the composite web weights in all three point design regions as shown in Table 20-12.

Table 20-13 presents a comparison of unit wing weights for the 3 point design regions for the chordwise stiffened hat section design and the monocoque honeycomb sandwich design. The unit weights for the surface panels and individual substructure components are shown. The minimum weight design for each point design region is identified by the cross-hatching. Trends similar to the metallic design are noted, with the chordwise stiffened being least weight for the lightly loaded forward box region (40322) and the honeycomb design being least weight for the highly loaded aft box and stiffness critical tip box structure. These unit weights are applied to establish the total wing weights for the advanced technology aircraft.

Results - Fuselage

The assessment of the potential payoff for composite technology application to the primary shell structure was made observing practical constraints for passenger accomodation. The two major factors included: (1) the need for passenger windows and (2) the requirement for ingress and emergency egress.

It is foreseeable to arrive at a "windowless" aircraft with an aesthetic mural for the interior design for an advanced technology supersonic transport. This would provide further opportunities to exploit fully the sandwich shell design or shell

TABLE 20-10. CIRCULAR-ARC CORRUGATED SPAR/RIB WEBS - REFERENCE TITANIUM ALLOY

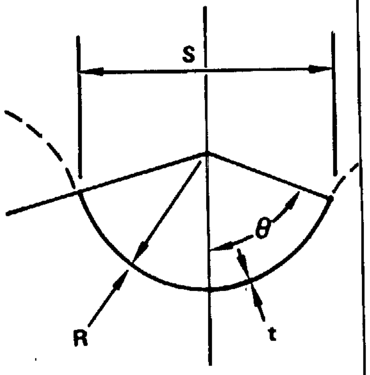
POINT DESIGN REGION	SPAC.	t	R	θ	S	A	$\rho\bar{t}$	$E\bar{t}$	$G\bar{t}$	w	GEOMETRY
	(in.)	(in.)	(in.)	(deg)	(in.)	(in^2)	(lb/in^2)	(10^6 lb/in.)	(10^6 lb/in.)	(lb/ft^2)	
40322											 TYPICAL WEB CROSS-SECTION
SPAR WEB	30	.022	1.30	80	2.56	.080	.0050	0.50	0.14	1.031	
RIB WEB	60	.025	1.50	77	2.92	.101	.0055	0.55	0.16	0.570	
40536											
SPAR WEB	20	.025	1.40	66	2.56	.081	.0051	0.51	0.16	0.980	
RIB WEB	60	.036	1.40	70	2.63	.123	.0058	0.58	0.22	0.486	
41348											
SPAR WEB	30	.027	1.00	60	1.732	.056	.0052	0.52	0.17	0.300	
RIB WEB	60	.018	1.00	60	1.732	.038	.0029	0.29	0.11	0.100	

TABLE 20-11. COMPOSITE CIRCULAR-ARC CORRUGATED SPAR WEB - POINT DESIGN 40322

POINT DESIGN REGION	REFERENCE (T_1)			BORON/POLYIMIDE (B/PI)					GRAPHITE/POLYIMIDE (GR/PI)				
	$E\bar{t} \times 10^6$ (lb/in^2)	$G\bar{t} \times 10^6$ (lb/in^2)	w_{T_1} (lb/ft^2)	$\bar{t}$ (in.)	$\rho\bar{t}$ (lb/in^2)	$E\bar{t} \times 10^6$ (lb/in^2)	$G\bar{t} \times 10^6$ (lb/in^2)	w_{B}/PI (lb/ft^2)	$\bar{t}$ (in.)	$\rho\bar{t}$ (lb/in^2)	$E\bar{t} \times 10^6$ (lb/in^2)	$G \cdot \bar{t} \times 10^6$ (lb/in^2)	w_{Gr}/PI (lb/ft^2)
40322 SPAR WEB	0.50	0.14	1.031	.053	.00379	0.80	0.19	0.773	.067	.00376	0.74	0.17	0.763

TABLE 20-12. COMPOSITE CIRCULAR-ARC CORRUGATED WEB WEIGHTS

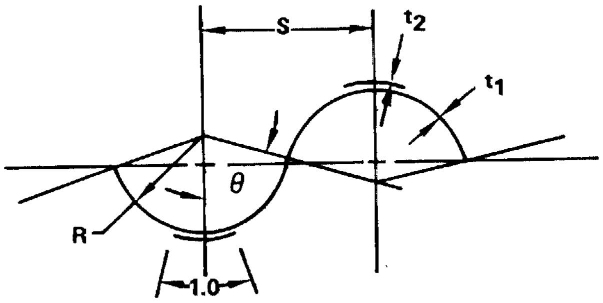
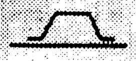
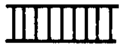
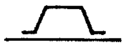
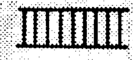
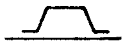
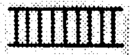
GEOMETRY	POINT DESIGN REGION	CIRCULAR-ARC CORRUGATION	REFERENCE Ti - 6A1 - 4V (lb/ft^2)	BORON-POLYIMIDE (lb/ft^2)	GRAPHITE-POLYIMIDE (lb/ft^2)
 $S = 2R \cos(90 - \theta)$ $A_1 = 2R\theta t_1; A_2 = 1.0t_2$ $\bar{t} = \Sigma An/S$	40322	SPAR WEB RIB WEB	1.031 0.570	0.773 0.428	0.763 0.422
	40536	SPAR WEB RIB WEB	0.980 0.486	0.735 0.365	0.725 0.360
	41348	SPAR WEB RIB WEB	0.300 0.100	0.227 0.076	0.226 0.075

TABLE 20-13. WING WEIGHT COMPARISON FOR POINT DESIGN REGIONS

POINT DESIGN REGION	40322				40536				41348			
STR. ARRANGEMENT	CHORDWISE		MONOCOQUE		CHORDWISE		MONOCOQUE		CHORDWISE		MONOCOQUE	
SPAR SPACING (in.)	30	30	30	30	30	30	30	30	30	30	30	30
COMPOSITE MATERIAL	B/PI	GR/PI	B/PI	GR/PI	B/PI	GR/PI	B/PI	GR/PI	B/PI	GR/PI	B/PI	GR/PI
PANEL CONCEPT												
UPPER SURFACE	0.998	0.988	1.044	1.037	1.694	1.674	2.255	2.226	1.457	1.443	2.125	2.118
LOWER SURFACE	0.998	0.988	0.957	0.950	1.412	1.394	2.623	2.584	1.003	0.993	1.993	1.985
TRUSS SPAR	0.336	0.338	0.265	0.267	0.656	0.663	0.517	0.523	–	–	–	–
CORRUGA. SPAR	0.540	0.535	0.773	0.763	0.716	0.711	0.735	0.725	0.545	0.541	0.227	0.226
TRUSS RIB	0.102	0.095	0.359	0.362	0.099	0.100	0.349	0.354	–	–	–	–
CORRUGA. RIB	0.584	0.579	0.428	0.422	0.798	0.792	0.365	0.360	0.402	0.399	0.076	0.075
SPAR CAP	0.337	0.334	0.178	0.177	3.795	3.761	0.252	0.249	3.584	3.552	0.181	0.179
RIB CAP	0.099	0.098	0.138	0.137	0.118	0.117	0.130	0.129	0.118	0.117	0.131	0.130
Σ TOTAL (lb/ft^2)	3.994	3.955	4.142	4.115	9.288	9.212	7.226	7.150	7.109	7.045	4.733	4.713

NOTE: SPAR SPACING = 30 inch
RIB SPACING = 60 inch

structure optimization without a frame spacing constraint. However, to obtain the design trends for this study, constraints for frame spacing of 20-inches and frame height of 3.0-inches were observed. Furthermore, the aforementioned constraints are consistent with the titanium skin-stringer and frame design evaluated in Task I and a direct comparison can be made to relate more directly the impact of composite utilization on the primary shell structure design.

The fuselage of the supersonic cruise aircraft is bending critical over most of its length as depicted in Figure 20-12, with internal pressure dictating requirements for the shell structure design forward of FS 1000. The basic concepts employed to satisfy the design requirements are shown in Figure 20-7. The tee-stringer design was adopted for the lightly loaded pressure critical forebody structure. For the bending critical regions both the tee-stringer and hat-stringer designs were evaluated.

For analysis, the fuselage shell was idealized as a circular shell as shown in Figure 20-15. The figure further reveals the skin-stringer and frame lamina directions identified in the subsequent tables. The analysis results of the fuselage cross section are identified by the notation of Figure 20-16; location 7 being representative of the lower centerline of the fuselage.

Tables 20-14 through 20-18 presents the sizing results of the design concepts employing boron-polyimide and graphite-polyimide composites. For comparison, the reference titanium shell properties are also indicated. In all cases the $\overline{Et}$ is greater for the composite material systems.

For fuselage frame design both the I-section and channel section frames were investigated. Typical frame-stringer intersections are shown in Figure 20-17. The figures indicate full depth frames with skin-flange continuity provided by integral clips. Table 20-19 indicates the proposed channel frames and the idealized frame employed for analysis. For composite application, a 10- to 20-percent increase in stiffness is indicated at FS 2000 and FS 3000 with a corresponding reduction in frame unit weights of approximately 20-percent, as shown in Table 20-20. To determine the frame weights at FS 2500 the above weight trend was applied and results tabulated.

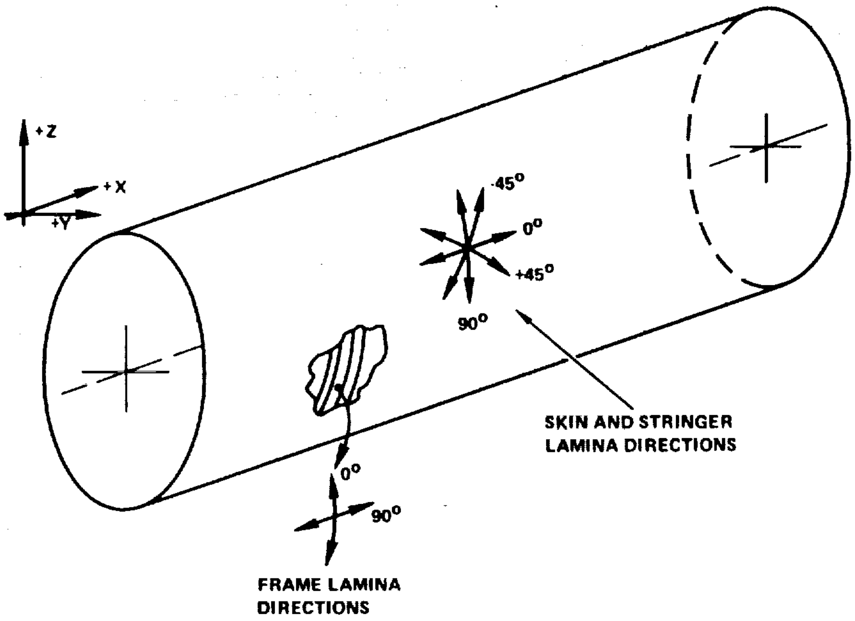

+Z
+X
+Y
-45°
0°
+45°
90°
SKIN AND STRINGER
LAMINA DIRECTIONS
0°
90°
FRAME LAMINA
DIRECTIONS

Figure 20-15. Composite Fuselage Laminate Directions

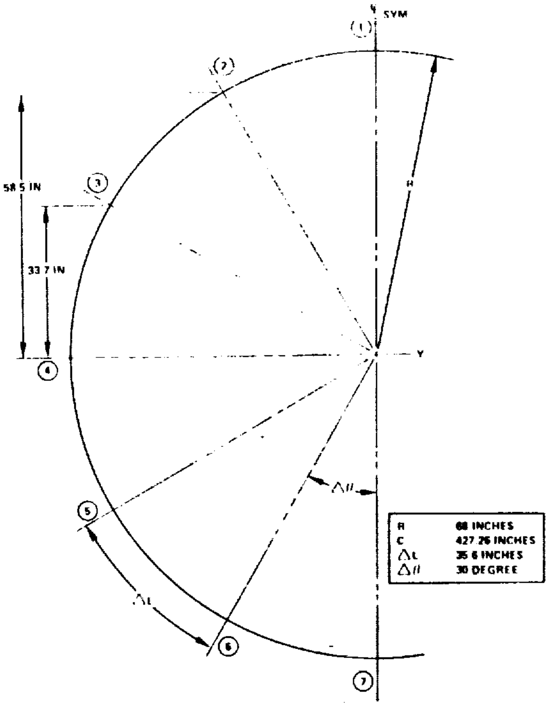
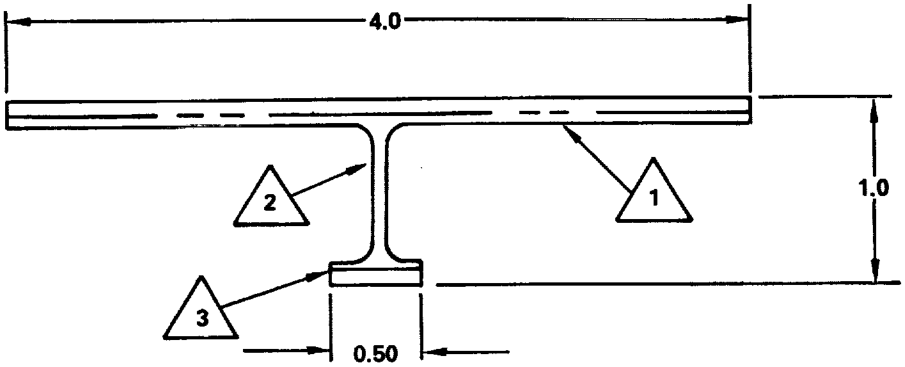

℄ SYM
①
②
③
④
⑤
⑥
⑦
58.5 IN
33.7 IN
R
Y
Δθ
Δℓ
R 68 INCHES
C 427.26 INCHES
Δℓ 35.6 INCHES
Δθ 30 DEGREE

Figure 20-16. Idealized Cross-Section of Fuselage

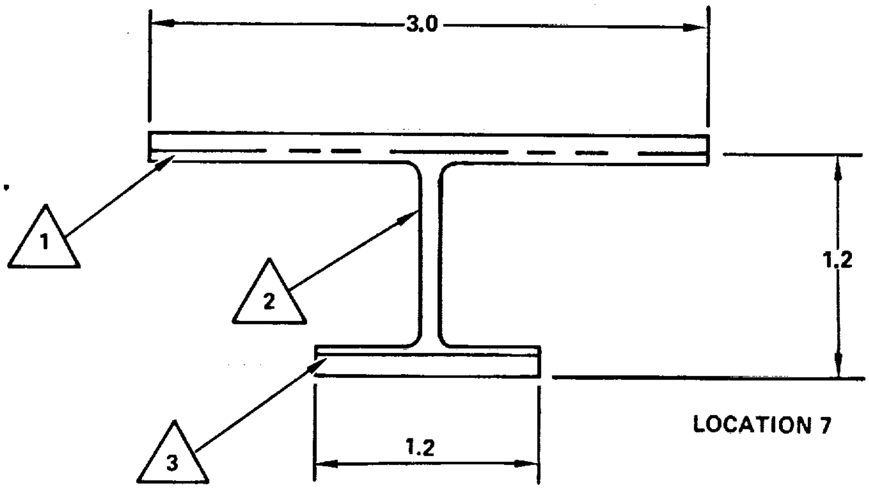
TABLE 20-14. COMPOSITE TEE-STIFFENED PANEL - FS 750



BORON-POLYIMIDE				GRAPHITE-POLYIMIDE			
NO.	ORIENTATION	b(in)	t(in)	NO.	ORIENTATION	b(in)	t(in)
1	$0_8/\pm45_4/90_4$	–	0.088	1	$0_8/\pm45_4/90_4$	–	0.112
2	$\pm45_4$	0.90	0.022	2	$\pm45_4$	0.90	0.028
3	$0_{12}/\pm45_4$	0.25	0.088	3	$0_{12}/\pm45_4$	0.25	0.112
$E\bar{t} \times 10^6$ (lb/in)		2.10		1.92			
$\rho\bar{t}$ (lb/in^2)		.00756		.00745			

REFERENCE TITANIUM: $E\bar{t} = 0.92 \times 10^6$ lb/in; $\rho\bar{t}$ = .009 lb/in^2

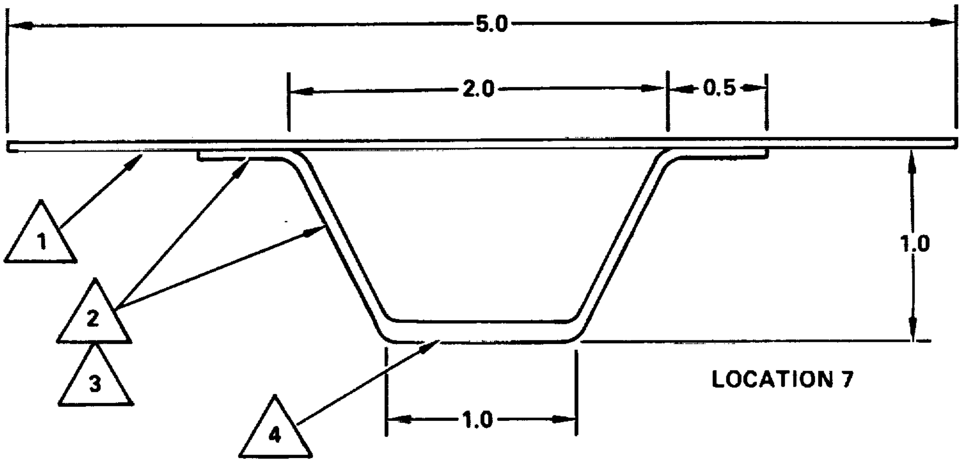
TABLE 20-15. COMPOSITE TEE-STIFFENED PANEL - FS 2000 AND FS 3000



BORON-POLYIMIDE				GRAPHITE-POLYMIDE			
NO.	ORIENTATION	b(in)	t(in)	NO.	ORIENTATION	b(in)	t(in)
1	$0_{12}/\pm45_{12}/90_4$	–	0.154	1	$0_{12}/\pm45_{12}/90_4$	–	0.196
2	$0_2/\pm45_{12}$	1.20	0.077	2	$0_2/\pm45_{12}$	1.20	0.098
3	$0_{16}/\pm45_{12}$	0.60	0.154	3	$0_{16}/\pm45_{12}$	0.60	0.196
$E\bar{t} \times 10^6$ (lb/in)		3.95		3.64			
$\rho\bar{t}$ (lb/in^2)		.0176		.0175			

REFERENCE TITANIUM: $E\bar{t} = 2.58 \times 10^6$ lb/in; $\rho\bar{t} = .0254$ lb/in^2

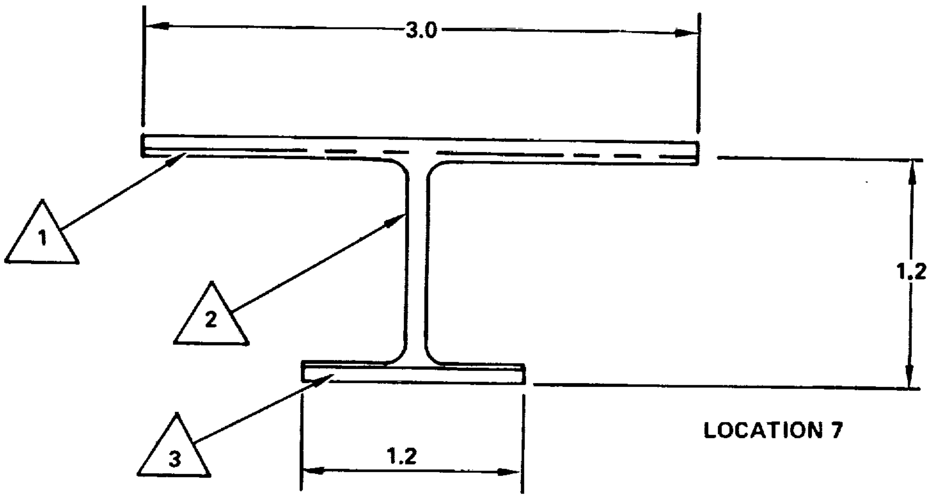
TABLE 20-16. COMPOSITE HAT STIFFENED PANEL - FS 2000 AND FS 3000



BORON-POLYIMIDE				GRAPHITE-POLYMIDE			
NO.	ORIENTATION	b(in)	t(in)	NO.	ORIENTATION	b(in)	t(in)
1	$0_{12}/\pm45_{12}/90_4$	–	0.154	1	$0_{12}/\pm45_{12}/90_4$	–	0.196
2	$\pm45_{12}$	1.15	0.066	2	$\pm45_{12}$	1.15	0.084
3	$\pm45_{12}$	1.15	0.066	3	$\pm45_{12}$	1.15	0.084
4	$0_{12}/\pm45_{12}$	1.00	0.132	4	$0_{12}/\pm45_{12}$	1.00	0.168
$E\bar{t} \times 10^6$ (lb/in)		3.14		2.91			
$\rho\bar{t}$ (lb/in^2)		.0161		.0160			

REFERENCE TITANIUM: $E\bar{t} = 2.58 \times 10^6$ lb/in; $\rho\bar{t} = .0254$ lb/in^2

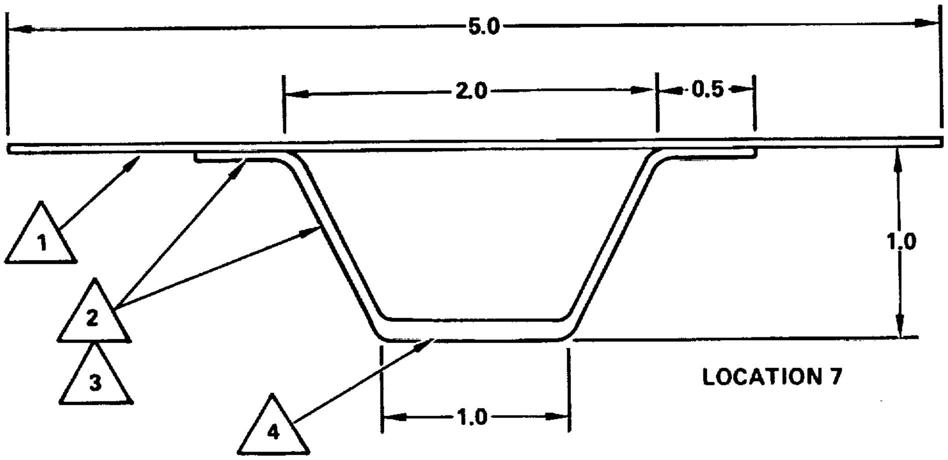
TABLE 20-17. COMPOSITE TEE-STIFFENED PANEL - FS 2500



BORON-POLYMIDE				GRAPHITE-POLYMIDE			
NO.	ORIENTATION	b(in)	t(in)	NO.	ORIENTATION	b(in)	t(in)
1	$0_{16}/\pm45_{12}/90_4$	–	0.176	1	$0_{16}/\pm45_{12}/90_4$	–	0.224
2	$0_2/\pm45_{12}$	1.20	0.077	2	$0_2/\pm45_{12}$	1.20	0.098
3	$0_{20}/\pm45_{12}$	0.60	0.176	3	$0_{20}/\pm45_{12}$	0.60	0.224
$E\bar{t} \times 10^6$ (lb/in)		4.94		4.53			
$\rho\bar{t}$ (lb/in^2)		.0200		.0202			

REFERENCE TITANIUM: $E\bar{t} = 3.06 \times 10^6$ lb/in; $\rho\bar{t} = .0302$ lb/in^2

TABLE 20-18. COMPOSITE HAT-STIFFENED PANEL - FS 2500



BORON-POLYIMIDE				GRAPHITE-POLYIMIDE			
NO.	ORIENTATION	b(in.)	t(in.)	NO.	ORIENTATION	b(in.)	t(in.)
1	$0_{16}/\pm45_{12}/90_4$	–	0.176	1	$0_{16}/\pm45_{12}/90_4$	–	0.224
2	$\pm45_{16}$	1.15	0.088	2	$\pm45_{16}$	1.15	0.112
3	$\pm45_{16}$	1.15	0.088	3	$\pm45_{16}$	1.15	0.112
4	$0_{16}/\pm45_{16}$	1.00	0.176	4	$0_{16}/\pm45_{16}$	1.00	0.224
$E\bar{t} \times 10^6$ (lb/in.)		4.04		3.73			
$\rho\bar{t}$ (lb/in.^{2})		0.0194		0.0192			

REFERENCE TITANIUM: $Et = 3.06 \times 10^6$ lb/in.; $\rho\bar{t} = 0.0302$ lb/in.^{2}

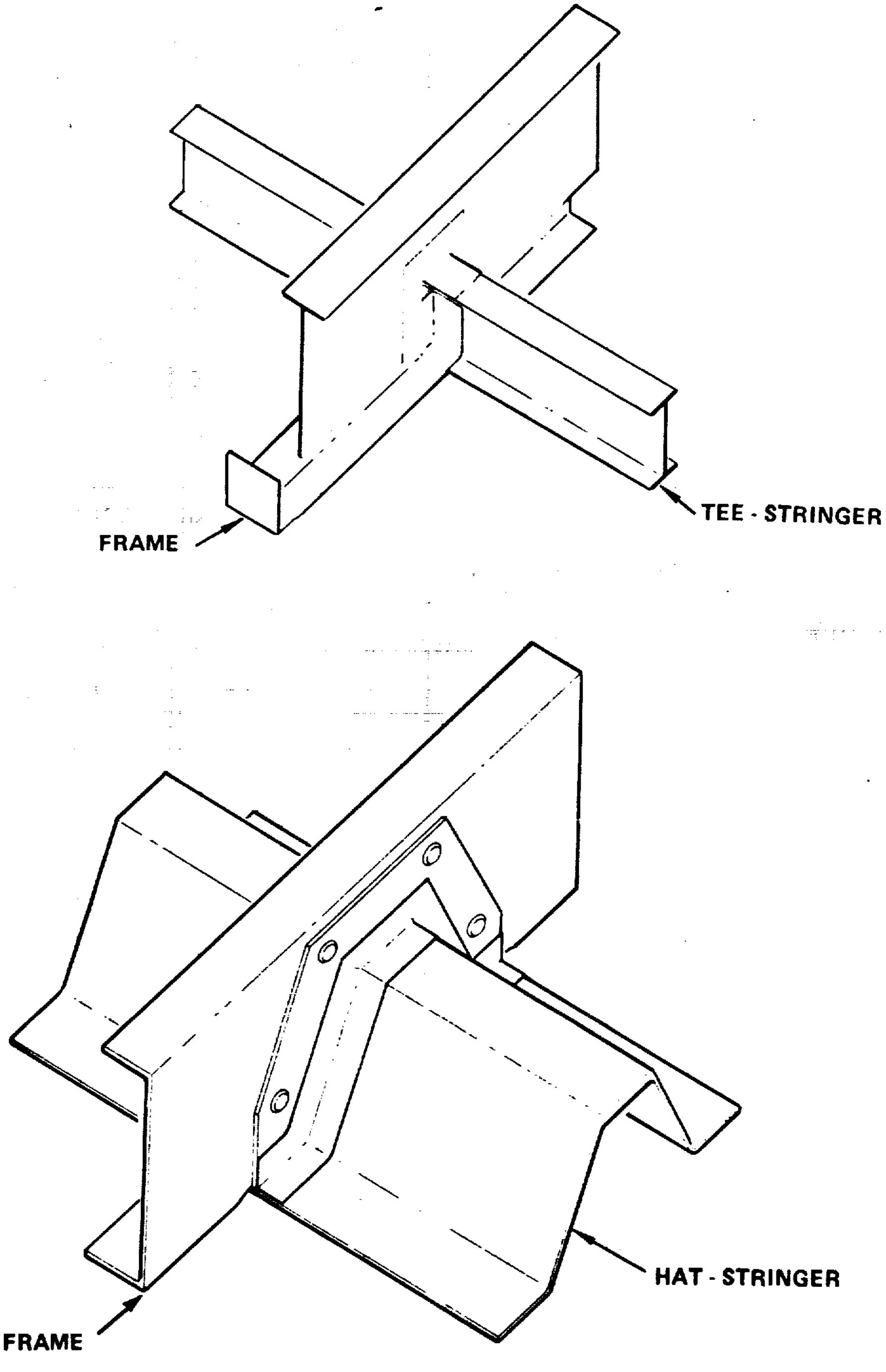
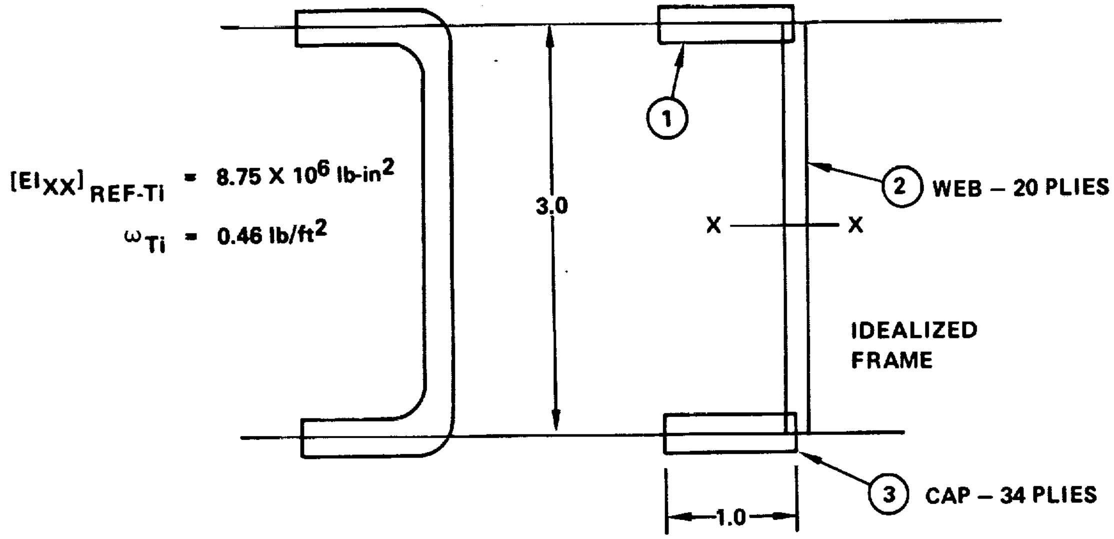

TEE - STRINGER
FRAME
HAT - STRINGER
FRAME

Figure 20-17. Typical Frame-Stringer Intersection Schemes

TABLE 20-19. COMPOSITE FUSELAGE FRAMES - FS 2000 AND FS 3000



$[EI_{XX}]_{REF\text{-}Ti} = 8.75 \times 10^6\ \text{lb-in}^2$

$\omega_{Ti} = 0.46\ \text{lb/ft}^2$

BORON-POLYIMIDE.

ELEMENT		AREA	ORIENTATION	E X 10^6	EA X 10^6
1	1.0 X 34 X 0.0055	0.187	$0_{18}/\pm45_{16}$	10.88	2.034
2	3.0 X 20 X 0.0055	0.330	$0_{4}/\pm45_{16}$	6.36	2.099
3	1.0 X 34 X 0.0055	0.187	$0_{18}/\pm45_{16}$	10.88	2.034
Σ		0.704			6.167

$$EI_{xx} = \left[(10.88 \times 10^6)(2)(0.187)(1.5)^2 + (6.36 \times 10^6)(0.110)(3)^3/12\right] = 10.73 \times 10^6\ \text{lb-in.}^2$$

$$\rho\bar{t} = (0.072)(0.704/20) = 0.00253\ \text{lb/in.}^2$$

GRAPHITE-POLYIMIDE.

ELEMENT		AREA	ORIENTATION	E X 10^6	EA X 10^6
1	1.0 X 34 X 0.0070	0.238	$0_{18}/\pm45_{16}$	7.60	1.809
2	3.0 X 20 X 0.0070	0.420	$0_{4}/\pm45_{16}$	4.74	1.991
3	1.0 X 34 X 0.0070	0.238	$0_{18}/\pm45_{16}$	7.60	1.809
Σ		0.896			5.609

$$EI_{xx} = \left[(7.60 \times 10^6)(2)(0.238)(1.5)^2 + (4.74 \times 10^6)(0.140)(3)^3/12\right] = 9.63 \times 10^6\ \text{lb-in.}^2$$

$$\rho\bar{t} = (0.056)(0.896/20) = 0.00251\ \text{lb/in.}$$

TABLE 20-20. COMPOSITE FRAME WEIGHTS

F.S. 2000 AND F.S. 3000

FRAME MATERIAL	EI_{xx} (10^6 lb-in.^{2})	EI_{xx} INCREASE (PERCENT)	$\rho\bar{t}$ (10^3 lb/in.^{2})	w (lb/ft^2)	WEIGHT SAVINGS (PERCENT)
TITANIUM	8.75	–	3.20	0.46	–
BORON-POLYIMIDE	10.73	22.6	2.53	0.365	20.6
GRAPHITE-POLYIMIDE	9.63	10.1	2.51	0.362	21.3

F.S. 2500

FRAME MATERIAL	EI_{xx} (10^6 lb-in.^{2})	EI_{xx} INCREASE (PERCENT)	ASSUMED WT SAVINGS (PERCENT)	w (lb/ft^2)
TITANIUM	10.94	–	–	0.576
BORON-POLYIMIDE	13.40	22.5	20.6	0.457
GRAPHITE-POLYIMIDE	12.05	10.1	21.3	0.452

The results of the fuselage skin panel and frame analysis are summarized on Table 20-21. The resulting weight trends for both boron-polyimide and graphite-polyimide composites are displayed for the panel concepts analyzed. Similar trends as observed for the metallic design are indicated with the tee-stiffener being least weight in the forebody and the hat stiffener design being least weight in the centerbody and aftbody structure. The data includes an estimate for the protection system weight of 0.045 lbs. per square foot. Also shown on the table is a summary of frame weight for the point design regions. The frame weights at FS 750 were conservatively taken as being equal to the requirements at FS 2000.

MANUFACTURING PLAN

The principal assumption for producing the 1980-1990 advanced technology aircraft is that polyimide resin systems will have developed to a point such that processing can be accomplished with ease. Thus, the low cost manufacturing method now being developed for epoxy processing were taken as feasible for polyimides. Restrictions on such factors as laminated thickness, bond pressure, etc. were neglected for this study.

Fabrication of ribs and spar caps as well as truss webs is accomplished by closed mold processing with elastomeric tooling as a pressure generator (Reference 5). Single stage molding and attachment of caps to truss or corrugated webs would be performed by similar techniques.

Wing skin panels, honeycomb or hat stiffened, would be produced with large sheets of material laminated by automated machines. Unit panels having dimension 10 ft. x 20 ft. were assumed. Since, for the most part, the wing skin gages are small, it assumed that the hat stiffener would first be produced as trapezoidal corrugation molded from a flat sheet. The hats would then be cocured to the skins using removable expansion mandrels. Because of contour complexity, flexible elastomeric tooling would be used extensively. The manufacturing sequence is pictorially displayed on Figure 20-18.

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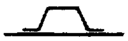
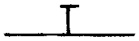
TABLE 20-21. COMPOSITE FUSELAGE WEIGHT SUMMARY

FUSELAGE SKIN PANELS					
MATERIAL SYSTEM	REFERENCE TITANIUM	BORON-POLYIMIDE	GRAPHITE-POLYIMIDE	BORON-POLYIMIDE	GRAPHITE-POLYIMIDE
w, UNIT WEIGHT	lb/ft^2	lb/ft^2	lb/ft^2	lb/ft^2	lb/ft^2
F.S. 750	1.29	–	–	1.134	1.118
F.S. 2000	2.74	2.363	2.349	2.579	2.565
F.S. 2500	3.02	2.839	2.810	2.925	2.954
F.S. 3000	2.90	2.363	2.349	2.579	2.565

$w_n = [(\rho \bar{t} \times 144) + (0.045)]$

FRAMES			
MATERIAL SYSTEM	REFERENCE TITANIUM	BORON-POLYIMIDE	GRAPHITE-POLYIMIDE
w, UNIT WT.	lb/ft^2	lb/ft^2	lb/ft^2
F.S. 750	0.25	0.365	0.362
F.S. 2000	0.53	0.365	0.362
F.S. 2500	0.576	0.457	0.452
F.S. 3000	0.53	0.365	0.362

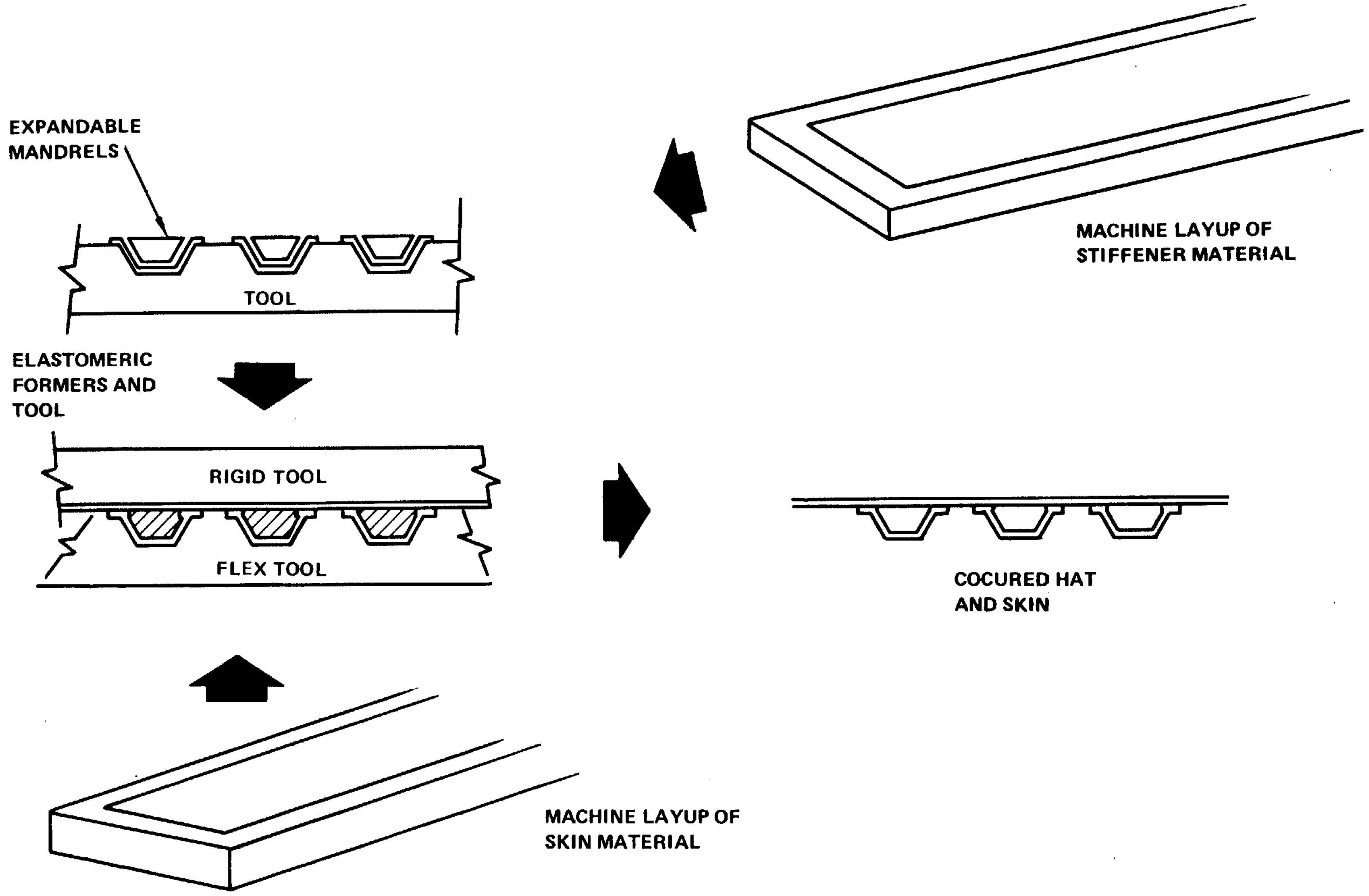

EXPANDABLE MANDRELS
TOOL
ELASTOMERIC FORMERS AND TOOL
RIGID TOOL
FLEX TOOL
MACHINE LAYUP OF SKIN MATERIAL
MACHINE LAYUP OF STIFFENER MATERIAL
COCURED HAT AND SKIN

Figure 20-18. Chordwise Stiffened Wing Panel Fabrication

MASS ESTIMATES

The advanced technology assessment exploited an aggressive application of composite materials to the airframe of a far-term supersonic transport. The weight advantages gained through the use of boron-polyimide (B/PI) and graphite-polyimide (Gr/PI) composites in the wing box and fuselage primary shell were determined through analysis of selected point design regions. The results of the advanced technology transport (ATT) studies (Reference 1) were used to establish weight trends for the other items. The comparison of weight trends for the far-term advanced technology supersonic transport and the near-term aircraft defined through indepth studies are summarized in Table 20-22.

A significant improvement in the fuel fraction for the fixed-size and weight airplane is shown for an all-composite and a hybrid far-term design. The range is increased from 4183 nautical miles to an excess of 4600 nautical miles, holding the payload constant at 49,000 pounds. When this aircraft carries a space-limit payload of 61,800 pounds (234 passengers x 200 + 1500 x 10) the range is approximately 4400 nautical miles. It is noted, however, that the fuel quantity indicated on the table exceeds the design capacity of 399,000 pounds. Thus, modification to the tank arrangement and capacity are required before a viable aircraft with the aforementioned performance potential can be achieved.

Another approach to exploit the weight advantages of composite application to the far-term design is resizing as shown in Table 20-22. The all-composite and hybrid design airplanes are resized to maintain a range of 4200 nautical miles with a payload of 49,000 pounds. The wing loading, takeoff thrust-to-weight ratio and fuel fraction are essentially held constant. For these cases, the taxi mass is 645,000 pounds and 641,500 pounds, for the all-composite and hybrid designs, respectively. The wing area has been reduced to approximately 9300 square feet. As indicated on the table, the Manufacturer's Empty Weight (MEW) has been reduced from 301,513 pounds to 246,762 pounds for the resized aircraft. This reduction of approximately 18-percent would result in a commensurate reduction in flyaway cost.

It is interesting to note that the reduction in structural mass between the near-term all-metal design and the far-term hybrid design is 21-percent. This result is consistent with the data of Reference 1 for the factors derived for the

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TABLE 20-22. ADVANCED TECHNOLOGY AIRCRAFT MASS COMPARISON (LB)

	NEAR-TERM		FAR-TERM			
			ALL-COMPOSITE		HYBRID	
ITEM	TASK I INITIAL	TASK I HYBRID	FIXED SIZE	RESIZED	FIXED SIZE	RESIZED
WING	109,600	88,620	78,320	68,740	77,455	67,577
STRUCTURAL BOX		47,268	45,279	40,320	44,414	39,316
"FIXED" WT. (L.E., T.E. ETC.)		41,352	33,041	28,420	33,041	28,261
TAIL	11,340	11,340	8,845	8,170	8,845	8,127
FUSELAGE	41.000	42,688	36,721	36,721	36,721	36,721
SHELL		23,148	19,981			
"FIXED" WT. (FLT. STA., FAIRING, ETC.)		19,540	16,740			
LANDING GEAR	30,400	30,400	27,360	24,250	27,360	24,118
AIR INDUCTION	17,800	20,755	18,036	15,220	18,036	15,137
NACELLES	4,900	5,616	4,880	4,200	4,880	4,174
ENGINES	44,600	51,124	51,124	43,950	51,124	43,728
PROPULSION SYSTEMS	7,000	7,310	7,310	7,022	7,310	6,984
SURFACE CONTROLS	8,500	8,500	6,800	5,982	6,800	5,950
INSTRUMENTS	1,230	1,230	1,230	1,230	1,230	1,230
HYDRAULICS	5,700	5,700	5,700	4,900	5,700	4,875
ELECTRICAL	4,550	4,550	4,550	4,485	4,550	4,461
AVIONICS	1,900	1,900	1,900	1,900	1,900	1,900
FURNISHINGS AND EQUIPMENT	11,500	11,500	11,500	11,500	11,500	11,500
ECS	8,300	8,300	8,300	8,300	8,300	8,300
TOLERANCE AND OPTIONS	1,980	1,980	1,980	1,980	1,980	1,980
MEW	310,300	301,513	274,556	248,550	273,691	246,762
STD. AND OPER. EQ.	10,700	10,809	11,054	10,300	11,062	10,284
DEW	321,000	312,322	285,610	253,850	284,753	257,046
PAYLOAD	49,000	49,000	49,000	49,000	49,000	49,000
ZFW	370,000	361,322	334,610	307,850	333,753	306,046
FUEL	380,000	388,678	415,390	337,150	416,247	335,454
TAXI MASS	750,000	750,000	750,000	645,000	750,000	641,500
RANGE, n. mi.	4,025	4,183	4,630	4,200	4,644	4,200
WING AREA, ft^2	10,822	10,882	10,882	9,307	10,822	9,257
BODY LENGTH, ft	297	297	297	297	297	297
W/S, lb/ft^2	69.3	69.3	69.3	69.3	69.3	69.3
W_{FUEL}/W_{TO}	0.507	0.518	0.554	0.523	0.555	0.523

40-percent advanced materials and applying the factors to the Task I weights shown. When comparing similar data to the hybrid design (i.e. composite reinforced) the reduction is 14-percent.

Wing Mass

The general expression used to evaluate the relative weights of the wing box structure is based on the three wing point design regions discussed earlier. The equation, which includes appropriate factors that produce consistent results with the Detailed Concept Analysis of Task I, is as follows:

$$W_{Box} = \frac{S_{Box}}{K_{Box}} (4.5308\ w_{40322} + 2.0000\ w_{40536} + 1.0540\ w_{41348})(NOF)$$

where, S_{Box}, box planform area = 7165 ft^2

K_{Box} = 7.5532

w_n, point design region unit weight (lb/ft^2)

NOF, non-optimum factor = 1.26

The total box weight (variable) is presented in Table 20-23 for the individual boxes (i.e. forward aft, tip). For comparative purpose the results of the hybrid design, Section 17, Concept Evaluation and Selection, is also shown.

Evaluation of the wing box weights for the near-term and far-term designs indicate the weight advantage of the minimum gage titanium alloy beaded panels of the forward box as compared to an equivalent stiffness composite design of either boron-polyimide or graphite-polyimide. For the stiffness critical tip structure, however, the application of composites afford a significant weight saving. The flutter penalty for the composite design is determined by providing equivalent added torsional stiffness based on the specific stiffness (G/ρ) parameter for the respective material systems as shown in Figure 20-19.

TABLE 20-23. ALL-COMPOSITE WING BOX STRUCTURE MASS COMPARISON

ITEM		TASK II HYBRID (A)	ADVANCED TECHNOLOGY	
			B/PI	Gr/PI
START-OF-DESIGN		NEAR-TERM	FAR-TERM	
POINT DESIGN REGION				
40322	(lb/ft^2)	3.80(B)	3.99	3.96
40536	(lb/ft^2)	7.27	7.23	7.15
41348	(lb/ft^2)	5.50	4.73	4.71
WING BOX				
FORWARD	(lb)	20,580(B)	21,607	21,445
AFT	(lb)	17,384	17,283	17,092
TIP	(lb)	6,964	6,016	5,962
FLUTTER INCR	(lb)	2,340(C)	809(C)	780(C)
Σ TOTAL	(lb)	47,268	45,715	45,279

NOTES:
- (A) COMPOSITE REINFORCED SPAR CAPS; BEADED PANELS EXCEPT H/C SANDWICH TIP BOX.
- (B) SIGNIFICANT ADVANTAGE OF METAL SURFACE PANELS AND COMPOSITE REINFORCED SPAR CAPS.
- (C) SIGNIFICANT ADVANTAGE OF COMPOSITE SANDWICH APPLICATION TO THE TIP STRUCTURE; FLUTTER INCREMENT BASED ON G/ρ RELATIONSHIP ASSUMING $\pm 45^\circ$ LAYUP.

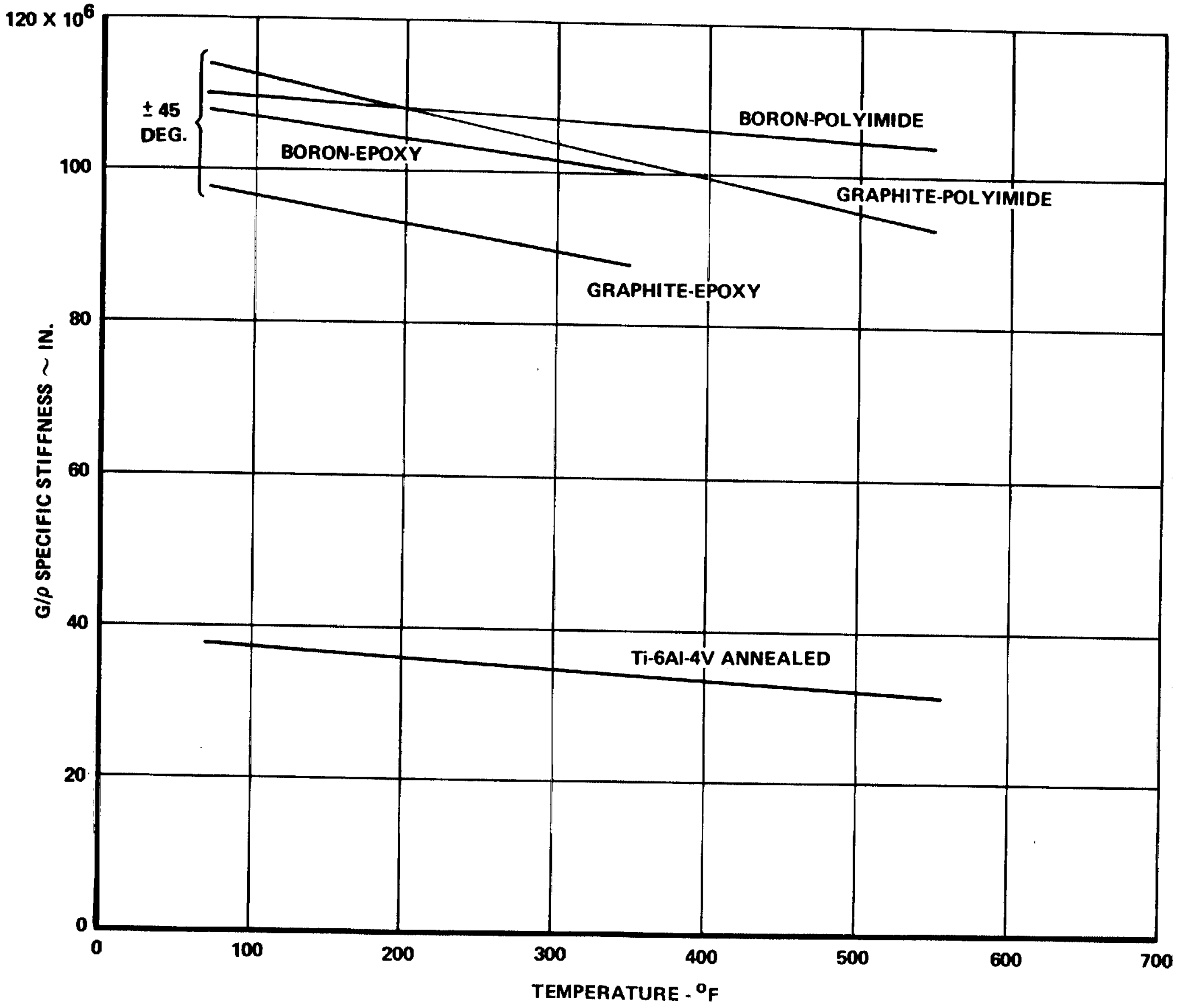

Figure 20-19. Specific Stiffness Properites of Titanium and Composites

Fuselage Mass

The general expression used to evaluate relative weights of the shell structure is based on the four point design region defined at F.S. 750, F.S. 2000, F.S. 2500 and F.S. 3000. The equation, which contains parameters defined in Section 15, is as follows:

$$W_{shell} = \frac{S_{shell}}{K_{shell}} (w_{750} + w_{2000} + w_{2500} + w_{3000})(NOF)$$

where, S_{shell}, shell wetted area = 7167 ft^2

K_{shell} = 4.1544

w_n, shell unit weight at each point design region (lb/ft^2)

NOF, non-optimum factor = 1.14

Table 20-24 presents the shell unit weights for each point design region and resulting total shell weight. Both boron/polyimide and graphite-polyimide material system data are shown along with corresponding weights for the all-titanium shell. A decrease in shell unit weight is reflected at all point design regions; the magnitude varies from a 4-percent to a 21-percent weight saving potential. A weight savings for the total shell when employing advanced composites is 14-percent.

Secondary Component Mass

The weight reduction factors for the secondary components were obtained from the results of the Advanced Technology Transport (ATT) studies conducted by the Lockheed-Georgia Company are reported in Reference 1. The appropriate reduction factors were applied to each component of the near-term design to arrive at the values used for the far-term aircraft as shown in Table 20-25. For the fixed weight items for the wing and fuselage, the overall reduction factor is 0.799 and 0.857, respectively.

The secondary components for the wing and fuselage total in excess of 60,000 pounds. The application of the respective reduction factors to the design results in a

MATERIAL SYSTEM:		NEAR-TERM	FAR-TERM		PERCENT CHANGE OVER NEAR-TERM
POINT DESIGN REGION	UNITS	TITANIUM 6A1-4V	BORON POLYIMIDE	GRAPHITE POLYIMIDE	
F.S. 750	$lb.ft^{-2}$	1.54	1.50	1.48	-3.9
F.S. 2000	$lb.ft^{-2}$	3.27	2.73	2.71	-17.1
F.S. 2500	$lb.ft^{-2}$	3.53	3.30	3.26	-7.6
F.S. 3000	$lb.ft^{-2}$	3.43	2.73	2.71	-21.0
W_{SHELL}	lb	23,148	20,178	19,981	-13.7

TABLE 20-24. ALL-COMPOSITE SHELL STRUCTURE MASS COMPARISON

TABLE 20-25. WEIGHT REDUCTION FACTORS FOR ADVANCED TECHNOLOGY AIRCRAFT SECONDARY COMPONENTS

ITEM	REDUCTION FACTOR(A)	NEAR-TERM	FAR-TERM
WING-FIXED WEIGHT			
LEADING EDGE	0.70	5,235	3,664
TRAILING EDGE	0.70	4,888	3,422
WING/BODY FAIRING	0.70	1,600	1,120
LEADING EDGE FLAPS	0.75	1,130	848
T.E. FLAPS	0.70	5,890	4,123
AILERONS	0.75	1,250	938
SPOILERS	0.75	1,360	1,020
MLG DOORS	0.75	2,904	2,178
SUPPORT STRUCTURE	1.00	3,750	3,750
B.L. 62 RIBS	0.86	1,430	1,230
FIN ATTACHMENT PROVISIONS	0.86	435	374
B.L. 470	0.86	700	602
REAR SPAR	0.86	3,400	2,924
ENGINE SUPPORT STRUCTURE	1.00	3,580	3,580
FUEL BULKHEADS	0.86	3,800	3,268
(SUBTOTAL)	0.799	41,352 lb	33,041 lb
FUSELAGE-FIXED WEIGHT			
NOSE AND FLIGHT STATION	0.93	2,500	2,325
NLG WELL	0.93	900	837
WINDSHIELD AND WINDOWS	1.00	1,680	1,680
FLOORING AND SUPPORTS	0.75	3,820	2,865
DOORS AND MECHANISM	0.80	4,170	3,336
UNDERWING FAIRING	0.70	1,870	1,309
CARGO COMP'T. PROV.	0.80	1,060	848
WING-TO-BODY FRAMES/FTGS.	1.00	1,500	1,500
TAIL-TO-BODY FRAMES/FTGS.	1.00	600	600
PROV. FOR SYSTEMS	1.00	740	740
FINISH AND SEALANT	1.00	700	700
(SUBTOTAL)	0.857	19,540 lb	16,740 lb

(A) REFERENCE 1

potential structural mass savings of approximately 10,000 pounds. These items alone offer significant weight payoff and improve aircraft performance for the supersonic cruise aircraft design.

FINAL DESIGN-ADVANCED TECHNOLOGY AIRCRAFT

The final design resulting from this advanced technology assessment is a hybrid structural approach as shown in Figure 20-20. The design makes extensive use of graphite/polyimide material system with a protective system of aluminum wire fabric and 120 glass. The chordwise stiffened structural arrangement-convex beaded surface panel concept of titanium alloy 6Al-4V resulted in minimum weight for the lightly loaded forward wing box structure. For the strength-stiffness critical wing aft box and tip structure, the honeycomb core sandwich using graphite/polyimide faces was found to be least weight. The fuselage structural arrangement is a skin-stringer-frame approach employing closed-trapezoidal hat stiffeners in the center body and aftbody with tee-section stiffeners used in the pressure critical forebody design.

The manufacturing plan postulated the applicability of low cost methods now being developed for epoxy processing. Fabrication of ribs and spar caps as well as truss webs would be accomplished by closed-mold processing with elastomeric tooling as the pressure generator. Single stage molding and attachment of caps to truss or corrugated webs would be performed by similar techniques. Wing skin panels would be produced with large sheets of material laminated by automated machines. Unit panels having dimensions of 10-feet x 20 feet are assumed. Because of contour complexity, flexible elastomeric tooling is extensively used.

The technology evaluation data presented in Table 20-26 compares the advanced technology aircraft with the near-term technology all-titanium and composite reinforced designs. The cascading effect of resizing of the advanced technology aircraft for a constant payload-range (49000-lb x 4200 n.mi) is shown. An 8.2-percent reduction in zero fuel weight considering a fixed airplane taxi weight is realized by the use of composites. This weight reduction is equated to a performance (range) increase of 500 nautical miles. Considering the growth factor

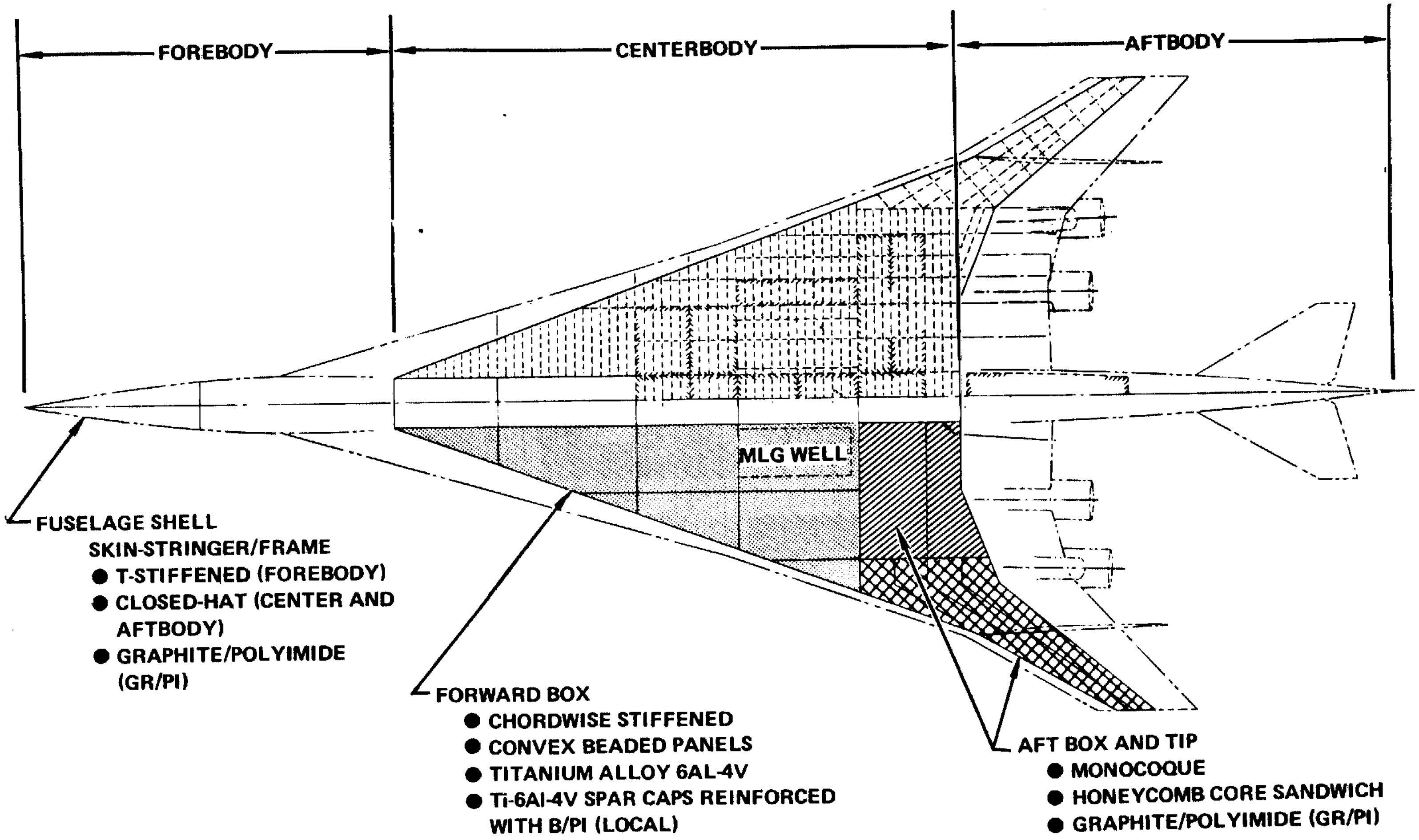
FOREBODY
CENTERBODY
AFTBODY
MLG WELL
FUSELAGE SHELL
SKIN-STRINGER/FRAME
● T-STIFFENED (FOREBODY)
● CLOSED-HAT (CENTER AND AFTBODY)
● GRAPHITE/POLYIMIDE (GR/PI)
FORWARD BOX
● CHORDWISE STIFFENED
● CONVEX BEADED PANELS
● TITANIUM ALLOY 6AL-4V
● Ti-6Al-4V SPAR CAPS REINFORCED WITH B/PI (LOCAL)
AFT BOX AND TIP
● MONOCOQUE
● HONEYCOMB CORE SANDWICH
● GRAPHITE/POLYIMIDE (GR/PI)

Figure 20-20. Advanced Hybrid Structural Approach - 1990 Start-of-Design (Far-Term)

TABLE 20-26. AIRPLANE MASS AND PERFORMANCE COMPARISON - ADVANCED TECHNOLOGY AIRCRAFT

		ADVANCED TECHNOLOGY	ADVANCED TECHNOLOGY	ADVANCED TECHNOLOGY	STRUCTURAL APPROACH TASK II ENGR. DESIGN STUDIES	NEAR-TERM TECHNOLOGY
TECHNOLOGY EVALUATION DATA		FIXED TAKE OFF WEIGHT	RESIZED CONSTANT RANGE –PAYLOAD	RESIZED CONSTANT RANGE –PAYLOAD		FIXED TAKE OFF WEIGHT
START-OF-DESIGN		FAR-TERM (1990)			NEAR-TERM (1980)	
PRIMARY MATERIAL SYSTEM		GRAPHITE/ POLYIMIDE	GRAPHITE/ POLYIMIDE	Ti-6Al-4V B/PI REINF. GRAPHITE/ POLYIMIDE	Ti-6Al-4V B/PI REINF.	Ti-6Al-4V
WING	STRUCTURAL ARRANGEMENT	HYBRID: CHORDWISE STIFFENED & MONOCOQUE				MONOCOQUE
	SURFACE PANEL CONCEPT	TRAPEZOIDAL CORRUGATION & HONEYCOMB CORE SANDWICH		CONVEX BEADED & H/C SANDWICH		H/C CORE SANDWICH
	STRUCTURAL BOX WEIGHT (lb) "FIXED" WT. (L.E., T.E., ETC.) (lb)	45,279 33,041	40,320 28,420	39,316 28,261	47,268 41,352	50,978 41,352
	TOTAL WEIGHT (lb)	78,320	68,740	67,577	88,620	92,330
FUSELAGE	STRUCTURAL ARRANGEMENT	SKIN-STRINGER-FRAME				
	SURFACE PANEL CONCEPT	TRAPEZOIDAL HAT STIFFENED		CLOSED HAT STIFFENED		
	SHELL WEIGHT (lb) "FIXED" WT. (FLT. STA., FAIRING, ETC.)	19,981 16,740	19,981 16,740	19,981 16,740	23,148 19,540	23,148 19,540
	TOTAL WEIGHT (lb)	36,721	36,721	36,721	42,688	42,688
ZFW, ZERO FUEL WEIGHT (lb)		334,610	307,850	306,046	361,322	364,982
GTW, GROSS TAXI WEIGHT (lb)		750,000	645,000	641,500	750,000	750,000
AIRPLANE RANGE (n.mi.)		4,630	4,200	4,200	4,183	4,123

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for this class of aircraft which consists of approximately 50-percent fuel, resizing of the airplane was accomplished. The airplane taxi weight was reduced by approximately 100,000 pounds.

The results of this assessment has identified the potential benefits of the composite materials and fabrication technology for application to a 1990-plus start-of-design Mach 2.7 supersonic cruise transport. The impact on the airplane size and weight are significant but require further in-depth analytical and experimental studies for validation. This includes, not only, the primary structural components, but also, secondary components including the leading edge, trailing edge, flight station, fairings, etc. which contribute significantly to the potential savings.

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3. E.B. Birchfield, F. Kollsmanberger, "Develop Fabrication/Processing Techniques for High Temperature Advanced Composites For Use in Aircraft Structures," AFMC-TR-72-91, McDonnell Aircraft Company, July 1972.

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SECTION 21

DESIGN METHODOLOGY

By

D. A. Brogan, G. W. Davis, G. A. Galins, E. A. Goforth, B. C. Wollner

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c.5

LIST OF SYMBOLS

b_o	Reference length
c	Local chord
C_{ℓ}	Local lift curve slope
$C_{L\alpha}$	Lift curve slope
$\overline{C}$	Area/span
$CC_{\ell}/\overline{C}C_{\ell\alpha}$	Normalized lift distribution
$\left[D_Z\right], \left[D_\theta\right]$	Deflection and slope transformation matrices
$\left[D_Z\right]^T, \left[D_\theta\right]^T$	Transpose of the deflection and slope transformation matrices
h	Height
$\left[K\right]$	Stiffness matrix
$\left[K_O\right]$	Base stiffness matrix
k	Reduced frequency, $\omega b_o/V$
M	Mach number
$\left[M\right]$	Mass matrix
M_X, M_Y	Bending moment about the x- and y-axis, respectively.
N_X, N_Y, N_{XY}	Normal and shearing forces per unit distance in the middle surface of plate
n	Number of values
n_Z	Vertical inertia load factor
P_Z	Force in the vertical direction
p	Non-dimensional time derivative operator, $\left(\frac{b_o}{V}\right)\left(\frac{d}{dt}\right)$
$\left[Q\right]$	Matrix of aerodynamic forces
t	Time; thickness
V	True airspeed
W	Weight
X, Y, Z	Cartesian coordinates
α	Angle of attack
β	Design variable
Δ	Incremental value(s)

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LIST OF SYMBOLS (Cont'd)

$\ddot{\theta}$	Pitching acceleration
$\ddot{\phi}$	Rolling acceleration
ω	Angular frequency in radians/second

SECTION 21

DESIGN METHODOLOGY

INTRODUCTION

To achieve the objectives established for this program, a systematic multidisciplinary analysis was conducted to assess the effects of the complete environment on the structural integrity of the aircraft. This analysis involved the complex interactions between static aeroelasticity, thermodynamics, flutter, static and dynamic loads, and strength. The flow diagram of the design cycle from initial definition of the airplane configuration to the establishment of the Final Design is presented in Figure 21-1. Due to the complex nature of this design cycle, extensive use of computer programs and their associated math models were required. These calculations were accomplished using Lockheed's Structural Design Analysis System, Figure 21-2, which is an integrated system with the combined program capabilities of the NASTRAN and the Lockheed FAMAS Systems.

The Lockheed FAMAS System contains a very extensive matrix algebra and manipulation system, and a large family of functional modules for aerodynamic loads, structural response and flutter analysis. In addition, this system has completely compatible matrix input/output capability within all its programs. NASTRAN provides finite element capabilities in statics, dynamics, and structural stability analyses for this system. These two systems are integrated with an interface module which permits transit from one system to the other. In this fashion all the design load analysis capabilities of the FAMAS system are joined with the finite element analysis capabilities of the NASTRAN System. Similarly, NASTRAN stiffness or structural flexibility matrices, vibration mode vectors, etc., can be used in direct link with the flutter analysis system in FAMAS, as well as the aeroelastic loads calculations.

This section describes the methodology involved with the diciplines of the analytical design cycle: Structural Temperature Analysis, Finite-Element Structural Analysis, Aeroelastic Loads Analysis, and the Vibration and Flutter Analysis. The methodology employed in the other disciplines are described in previous sections of this report.

STRUCTURAL TEMPERATURE ANALYSIS

A series of mathematical models was employed to describe the heat flow paths within the primary structure and for determination of transient temperature histories for the structural components. The principles of the procedure for prediction of aerodynamic heating and resulting structural temperatures for input to the structural design and analysis are displayed in Figure 21-3.

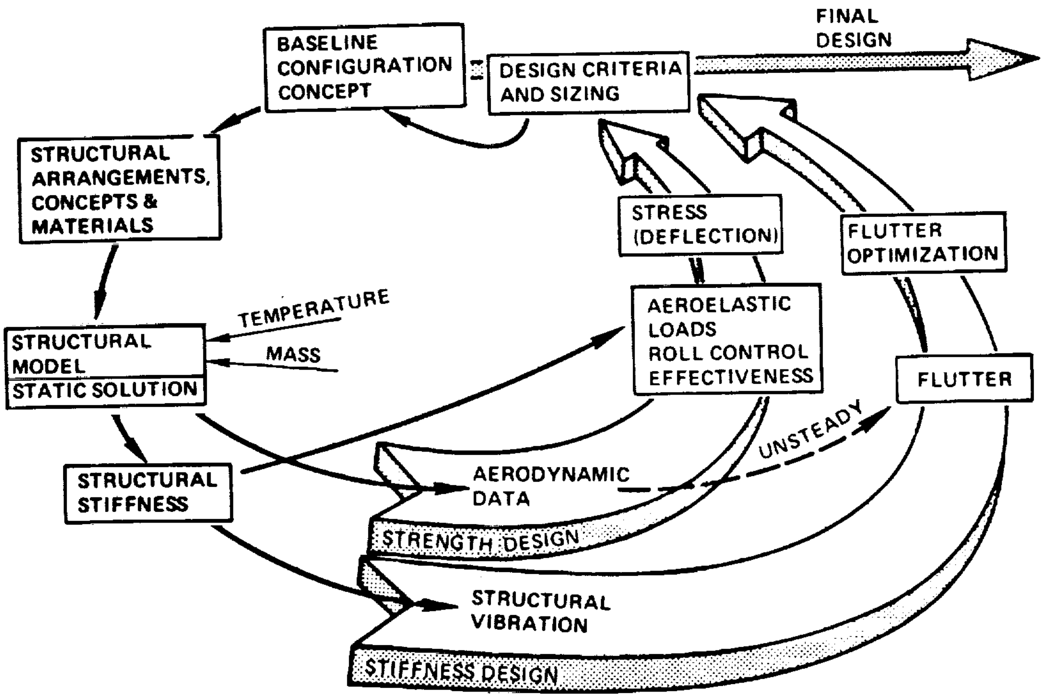
FINAL DESIGN
BASELINE CONFIGURATION CONCEPT
DESIGN CRITERIA AND SIZING
STRUCTURAL ARRANGEMENTS, CONCEPTS & MATERIALS
STRESS (DEFLECTION)
FLUTTER OPTIMIZATION
TEMPERATURE
MASS
STRUCTURAL MODEL
STATIC SOLUTION
AEROELASTIC LOADS ROLL CONTROL EFFECTIVENESS
FLUTTER
UNSTEADY
STRUCTURAL STIFFNESS
AERODYNAMIC DATA
STRENGTH DESIGN
STRUCTURAL VIBRATION
STIFFNESS DESIGN

Figure 21-1. Analytical Design Cycle

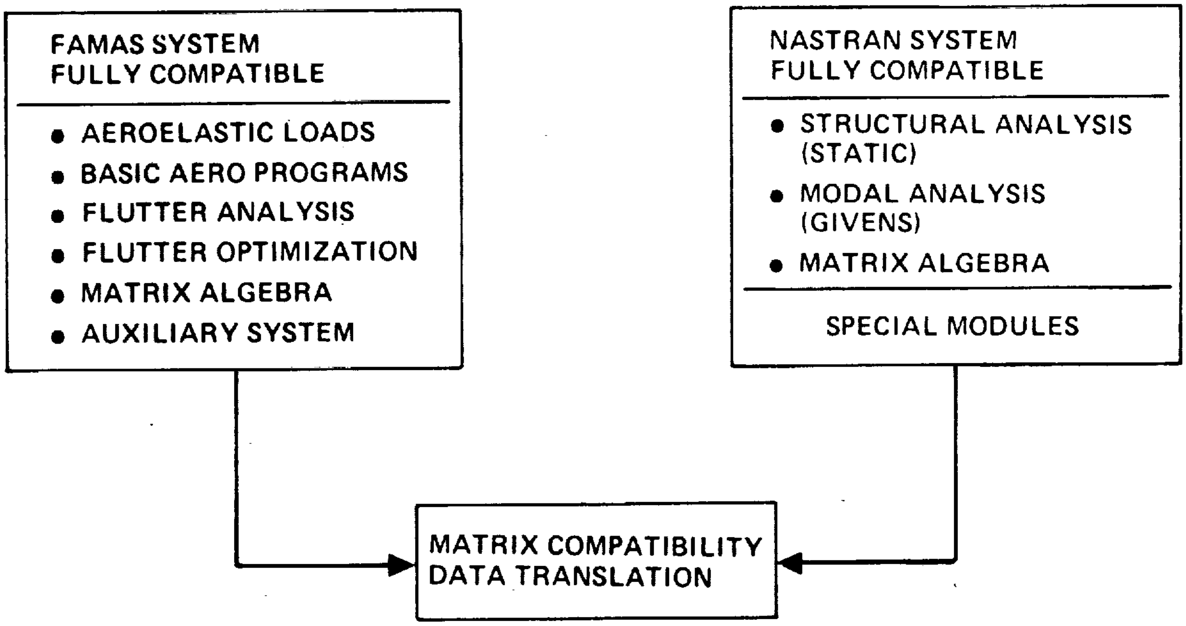
FAMAS SYSTEM FULLY COMPATIBLE
• AEROELASTIC LOADS
• BASIC AERO PROGRAMS
• FLUTTER ANALYSIS
• FLUTTER OPTIMIZATION
• MATRIX ALGEBRA
• AUXILIARY SYSTEM
NASTRAN SYSTEM FULLY COMPATIBLE
• STRUCTURAL ANALYSIS (STATIC)
• MODAL ANALYSIS (GIVENS)
• MATRIX ALGEBRA
SPECIAL MODULES
MATRIX COMPATIBILITY DATA TRANSLATION

Figure 21-2. Lockheed s Integrated Structural Design Analysis System

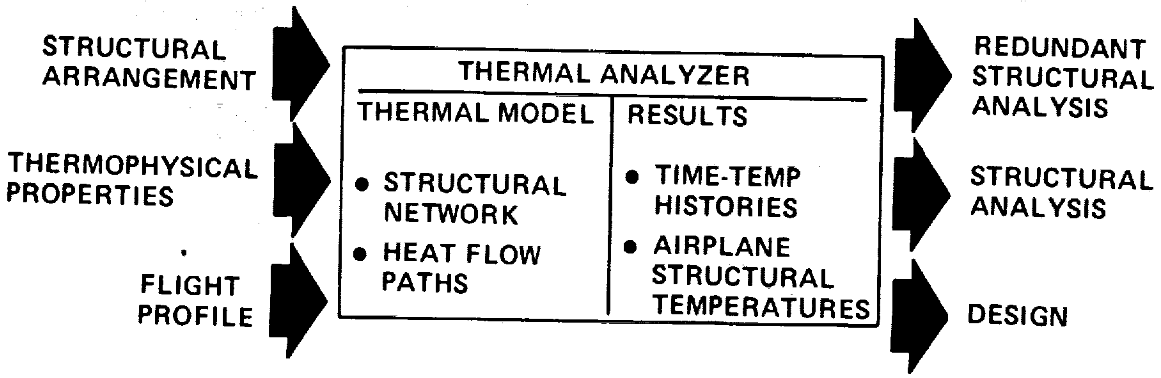
STRUCTURAL ARRANGEMENT
THERMOPHYSICAL PROPERTIES
FLIGHT PROFILE
THERMAL ANALYZER
THERMAL MODEL
• STRUCTURAL NETWORK
• HEAT FLOW PATHS
RESULTS
• TIME-TEMP HISTORIES
• AIRPLANE STRUCTURAL TEMPERATURES
REDUNDANT STRUCTURAL ANALYSIS
STRUCTURAL ANALYSIS
DESIGN

Figure 21-3. Evaluation Procedure for Structural Temperatures

Math Models

The mathematical models consist of networks of structural nodes interconnected by heat flow paths and set up for solution using Lockheed's Thermal Analyzer computer program (Reference 1). The solution method is analogous to the solution of voltage distribution in an electrical resistor-capacitor network: current (heat) flows through electrical (thermal) resistors as a function of the voltage (temperature) potential between connected points, and is stored as electrical (thermal) energy in components called capacitors (structural mass) at a rate that is a function of the electrical (heat) capacity of that component. Lockheed's Thermal Analyzer is a completely general and versatile computer program, permitting specification of any type of temperature- or time-dependent heat flow including conduction, convection, radiation, and variable heat storage. An additional capability allows reconnection of network elements during run time, permitting solution of complex problems such as exposure of fuel tank structure to interior radiation as fuel was drained from the tank.

The thermal networks were generalized to accept arbitrary dimensional data for applicability to similarly shaped structures. Detail dimensions were supplied as standard input data and the actual resistor and capacitor values calculated automatically for each case. This technique eliminated the need for minor network revisions each time a dimension was changed, and saved significant programming time.

All Thermal Analyzer networks were set up to compute in a transient mode. The flight profile for a "hot day" (standard plus 8K) international mission was used to determine aerodynamic heating and altitude effects. Cases were run from takeoff roll, to climb, through cruise, and descent to loiter before landing.

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Heat transfer in the interior of the wing was determined by setting up a wing box network. The network (Figure 21-4) includes sets of nodes for the upper and lower panels, plus one node each for the four vertical webs (to form a rectangular box). The shape of the box was determined by panel size (spanwise by chordwise dimensions) and by wing depth obtained from wing contour drawings. All node areas were normalized with reference to one square foot of panel surface area to facilitate resistor and capacitor calculations. Heat transfer within the wing box includes radiation exchange, convection to boundary layer air when leakage was a factor, and for fuel tank areas, convection to fuel and fuel vapor.

Two sets of nodal representations for the surface panels were derived for inclusion in the wing box network. The first set was for the corrugation or hat-section stiffened panel concepts, and the second set was for the honeycomb panels. An example of the node definitions for the honeycomb panels are shown in Figure 21-5. Nodes 2 and 3 are defined as the outer and inner halves of the core, respectively. Nodes 1 and 4 include thermal capacity of the braze material. Heat transfer within the panel includes conduction (nodes 1-2, 2-3, and 3-4) and radiation (nodes 1-2, 1-3, 1-4, 2-3, 2-4, and 3-4). To reduce network complexity and computer running time, all braze material was assumed to remain in contact with the face sheets.

Node definitions for the fuselage frame network are shown in Figure 21-6. This network was set up to determine the variation in average skin panel and frame temperatures around the circumference of the fuselage. The hat-section stiffened skin concept is shown, and skin-panel heat transfer is identical to that for type-1 wing panels. Heat transfer to the frame was by conduction and radiation from the skin panels, and by conduction from the surrounding insulation. Conduction through the insulation to the inner skin was also included. Boundary conditions on the inner surface of the fuselage wall included a low convection rate to cabin air and radiation to cabin interior. The network at the fuselage forebody station was adjusted to simulate zee-section stiffeners with the hat-section model.

Heat flow paths were defined by thermal resistors connected between nodes representing structure or between nodes and given boundary condition temperatures. Heat flows directly into a node were also defined explicitly. The Thermal Analyzer network included the following types of heat flow paths: external and internal convection, conduction, external and internal radiation, and thermal capacity.

Airframe Structural Temperatures

Structural temperatures were calculated to define the thermal environment at selected wing and fuselage locations for the detail stress analysis and on the overall configuration for input into the structural models, Figure 21-7. These temperatures were developed by using the thermal analysis networks and structural data of the arrow-wing configuration and performing the airplane over a nominal Mach 2.7 cruise flight profile.

Time-temperature histories were developed using the Thermal Analyzer computer program at 14 wing and 10 fuselage locations. These locations were selected to include those point design regions used for the detail stress

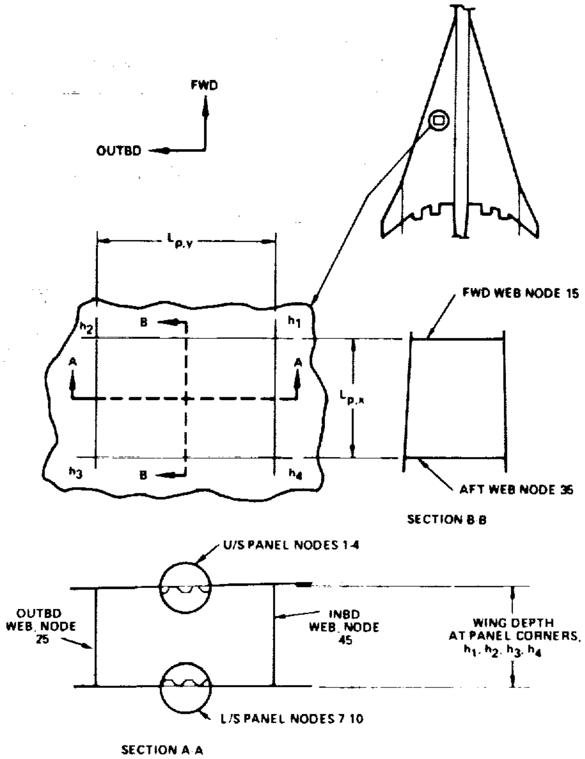
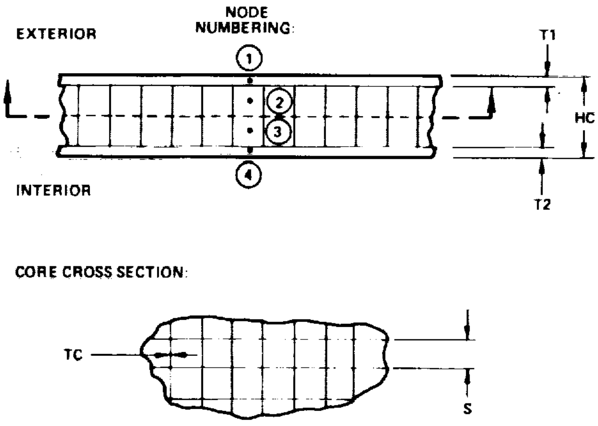

FWD
OUTBD
$L_{p,y}$
FWD WEB NODE 15
h_2
B
h_1
A
A
$L_{p,x}$
h_3
B
h_4
AFT WEB NODE 35
SECTION B-B
U/S PANEL NODES 1-4
OUTBD WEB NODE 25
INBD WEB NODE 45
WING DEPTH AT PANEL CORNERS, h_1, h_2, h_3, h_4
L/S PANEL NODES 7-10
SECTION A-A

Figure 21-4. Wing Box Thermal Analyzer Network


EXTERIOR
NODE NUMBERING:
T1
①
②
③
④
HC
INTERIOR
T2
CORE CROSS SECTION:
TC
S

DIMENSIONAL DATA:

HC	PANEL THICKNESS
T1	EXTERIOR FACE SHEET THICKNESS
T2	INTERIOR FACE SHEET THICKNESS
S	CORE CELL SIDE
TC	CORE FOIL THICKNESS

ADDITIONAL DATA:

WB	BRAZE WEIGHT/AREA

Figure 21-5. Network Node Definition, Type-2 Wing Panels

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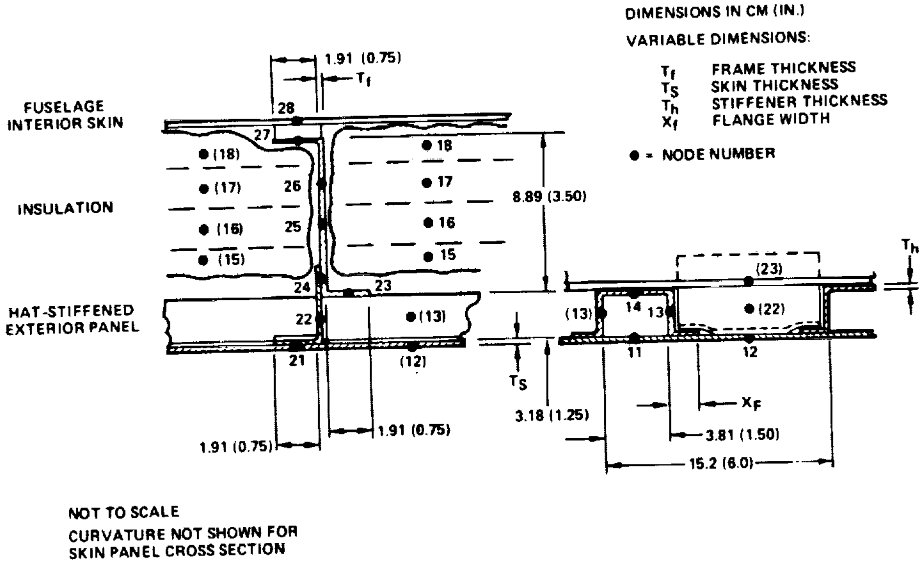
DIMENSIONS IN CM (IN.)
VARIABLE DIMENSIONS:
T_f FRAME THICKNESS
T_S SKIN THICKNESS
T_h STIFFENER THICKNESS
X_f FLANGE WIDTH
● = NODE NUMBER
FUSELAGE INTERIOR SKIN
INSULATION
HAT-STIFFENED EXTERIOR PANEL
1.91 (0.75)
T_f
28
27
● (18)
● (17)
● (16)
● (15)
26
25
24
23
22
21
● 18
● 17
● 16
● 15
● (13)
(12)
8.89 (3.50)
T_h
(23)
(13)
14
13
● (22)
11
12
T_S
3.18 (1.25)
X_F
3.81 (1.50)
15.2 (6.0)
1.91 (0.75)
1.91 (0.75)
NOT TO SCALE
CURVATURE NOT SHOWN FOR SKIN PANEL CROSS SECTION

Figure 21-6. Network Node Definition, Fuselage Panel and Frame

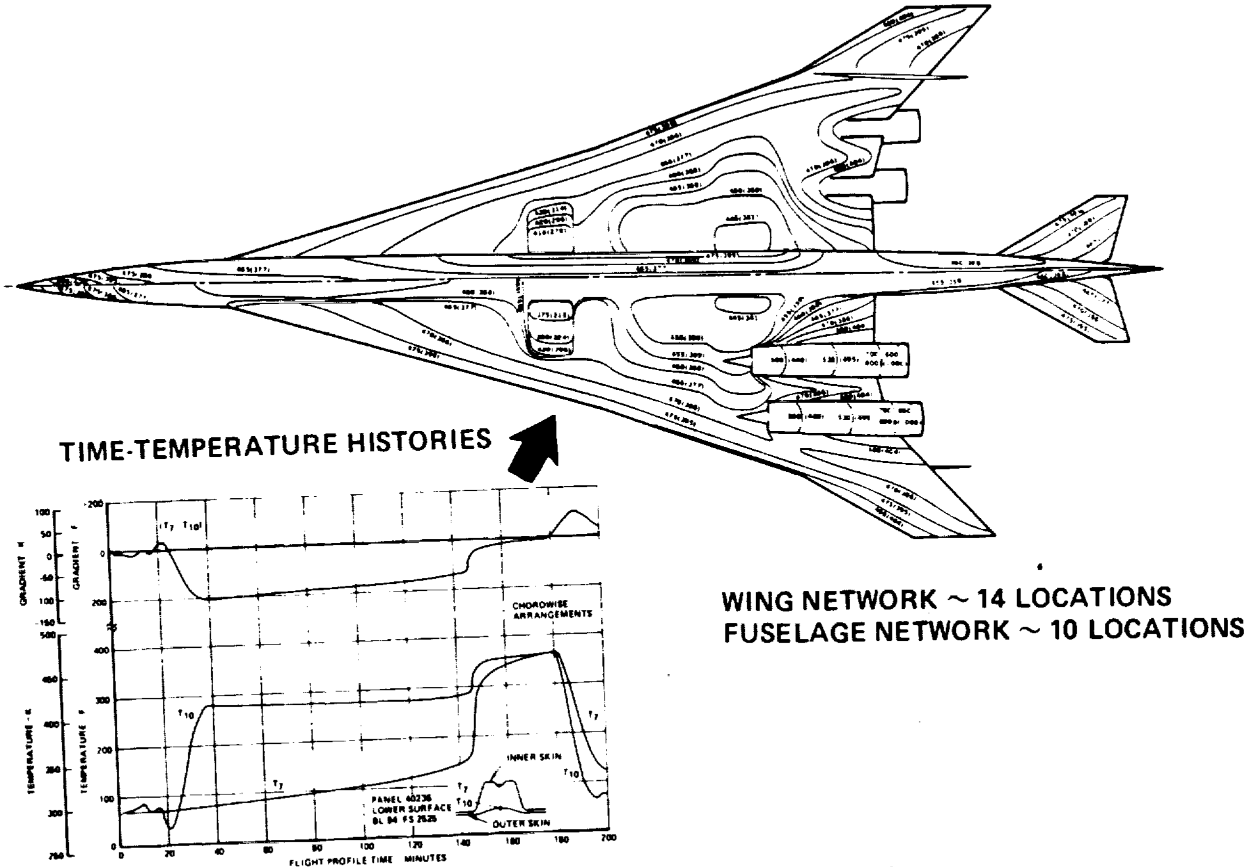

Figure 21-7. Airframe Structural Temperatures

analysis and provided the necessary temperature data to define the point design environment for these regions.

Temperatures were developed for the entire wing and fuselage based on the extrapolation of the time-temperature histories calculated at the selected wing and fuselage locations. In addition, grid point and element temperatures were developed from these overall wing and fuselage temperatures for inclusion in the finite-element structural model.

FINITE-ELEMENT STRUCTURAL ANALYSIS

A series of finite element (F.E.) structural analysis models were used for the evaluation of structural design concepts. These models were coded for NASTRAN, and Lockheed's Structural Design Analysis System was used to provide internal loads and displacements for stress analyses, to calculate structural deflection influence coefficients for aeroelastic load analyses, and to determine reduced stiffness and mass matrices and compute vibration modes for flutter analyses. The principles of this analytical procedure are shown in Figure 21-8, and generally consisted of the following steps:

(1) F.E. structural models were established using the basic airplane configuration and the flexibilities commensurate with the structural-material concept being studied. Plot routines and internal check runs were conducted to validate each model.

(2) NASTRAN redundant-structure analysis solutions were obtained using the F.E. models formulated in step (1). Plot routines were used to verify the continuity of the resulting displacement structural influence coefficients (SIC). Upon verification of the SIC, both SIC and stiffness matrices were released for the aeroelastic loads and vibration analyses, respectively. The vibration analysis initiated the vibration and flutter evaluation with no further interaction with the F.E. structural analysis until the next design iteration was attempted. The SIC matrix, in association with the basic aerodynamics, mass matrices, and the flight parameters was used to calculate the static aeroelastic loads.

(3) A NASTRAN internal loads run was conducted using the aeroelastic loads matrix derived in Step (2), with the resulting displacements and internal loads (stresses) subsequently used for the point design stress analysis.

Structural Models

The F.E. structural models used for the structural investigation are summarized on the flow schematic in Figure 21-9. These models are characterized by their basic modeling technique and their wing primary load-carrying structural arrangement. The three general types of structure and combination thereof evaluated were: chordwise-stiffened, spanwise-stiffened, biaxially-stiffened (monocoque), and the hybrid-stiffened arrangements. The two different modeling

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techniques employed in the formulation of these structural models were: the pseudo 2-Dimensional (2-D) method and the conventional 3-Dimensional (3-D) method. A statistical comparison of the two modeling techniques is as follows:

MODEL	GRID POINTS	ELEMENTS	DEGREES OF FREEDOM	CPU TIME* (HR)
2-D	530	1300	1050	0.37
3-D	715	2450	2200	0.72

* COMPUTER TIME TO GENERATE SYM. AND A/S SIC'S

In addition, the computer time required to formulate the symmetric and antisymmetric structural influence coefficients (SIC) are included and indicate the cost effectiveness of the pseudo 2-D model, i.e., the 2-D model is half as costly (computer run time) as the more detailed 3-D model.

The simplified pseudo 2-D model technique was formulated for the initial studies as a rapid, cost effective method for evaluating the effects of the primary wing loads (P_z, M_x and M_y). A representative description of the modeling technique used on the 2-D models is included in Figure 21-9. This model represents the actual wing upper surface planform with a simplified wing cross-section and the fuselage was idealized as a simple beam. For the wing, a horizontal midplane (X-Y plane) of structural symmetry was assumed, which permits the size of the model to be substantially smaller since only the upper half of the wing needs to be specified in the model. The wing cross-section of the model was symmetrical about the X-Y plane. The Z coordinates (measured from the X-Y plane) defined the upper wing surface and were equal to one-half of the total wing depth of the model. Section properties were equal to the average stiffness of the wing upper and lower surfaces.

For the wing vertical fin model, a two-dimensional (X-Z plane) grid system with NASTRAN bar and shear panel elements was used to represent the equivalent bending and torsion stiffness of the fin. Fin loads are introduced into the wing by means of NASTRAN multipoint constraint (MPC) equations which were applied at the interface of the fin with the wing box.

Engine support beams were represented in the model by NASTRAN bar elements with the capability to transmit axial, torsional, and vertical and lateral bending loads from the engines into the wing box. The beams were located at the constrained wing midplane and were connected by MPC equations to the X-Y rigid body motions of the wing for the vibration analyses.

The 2-D models represent the fuselage as a simple beam and used NASTRAN bar elements with torsional, and vertical and lateral bending stiffness. The fuselage beam was connected to the wing model by MPC equations and scalar springs (NASTRAN CELAS elements) representing an approximation of the fuselage frame flexibility.

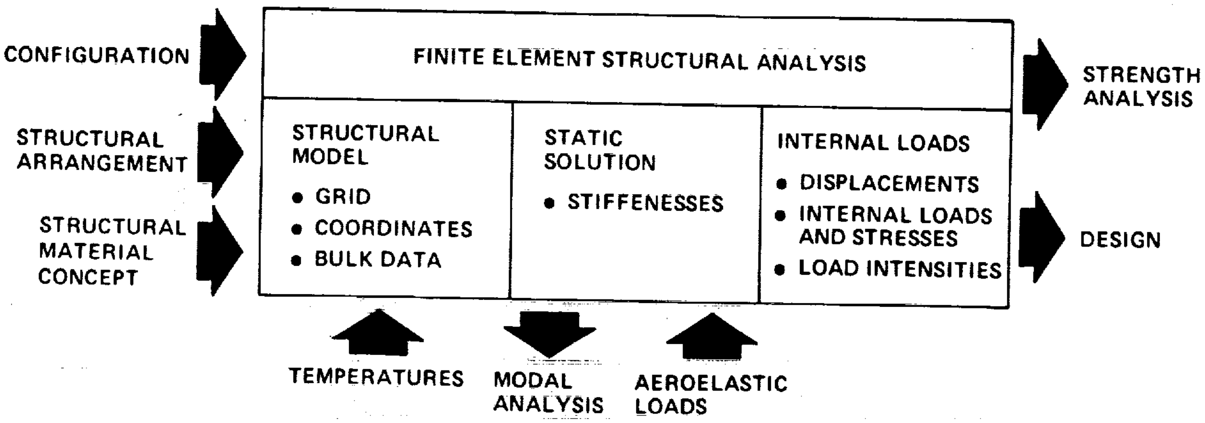
CONFIGURATION
STRUCTURAL ARRANGEMENT
STRUCTURAL MATERIAL CONCEPT
FINITE ELEMENT STRUCTURAL ANALYSIS
STRUCTURAL MODEL
• GRID
• COORDINATES
• BULK DATA
STATIC SOLUTION
• STIFFENESSES
INTERNAL LOADS
• DISPLACEMENTS
• INTERNAL LOADS AND STRESSES
• LOAD INTENSITIES
STRENGTH ANALYSIS
DESIGN
TEMPERATURES
MODAL ANALYSIS
AEROELASTIC LOADS

Figure 21-8. Finite Element Structural Analysis Procedure

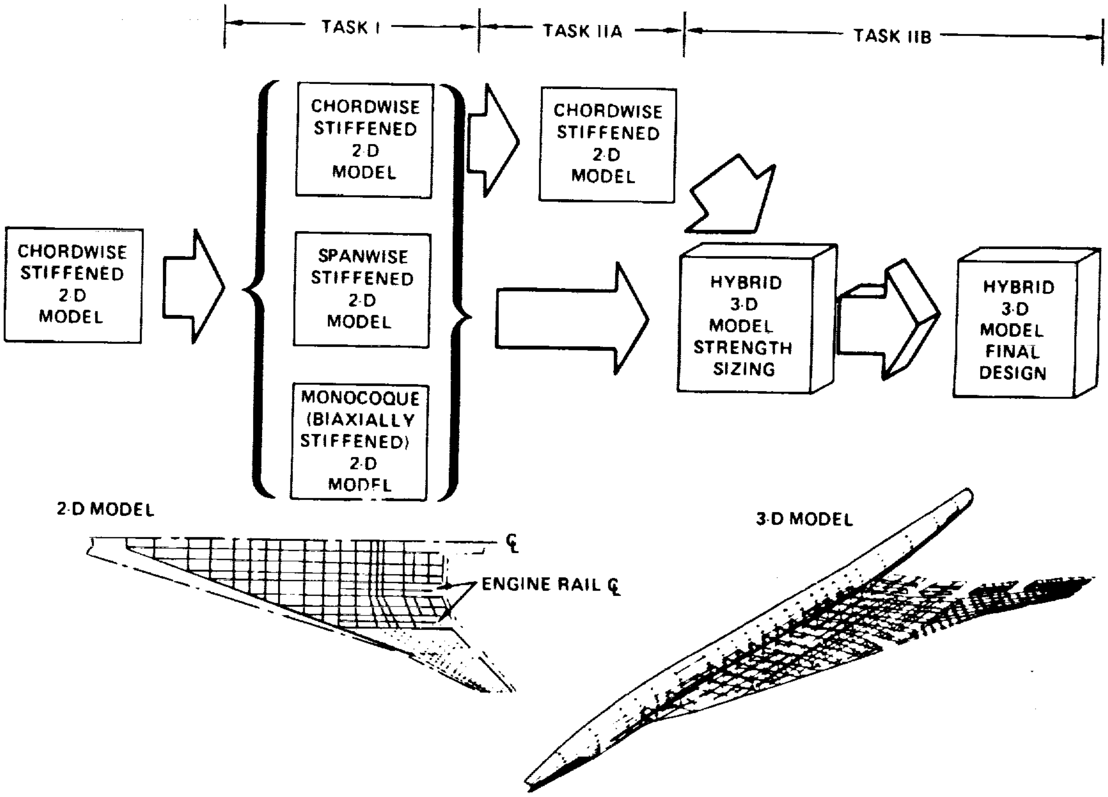
TASK I
TASK IIA
TASK IIB
CHORDWISE STIFFENED 2-D MODEL
CHORDWISE STIFFENED 2-D MODEL
SPANWISE STIFFENED 2-D MODEL
MONOCOQUE (BIAXIALLY STIFFENED) 2-D MODEL
CHORDWISE STIFFENED 2-D MODEL
HYBRID 3-D MODEL STRENGTH SIZING
HYBRID 3-D MODEL FINAL DESIGN
2-D MODEL
℄
ENGINE RAIL ℄
3-D MODEL

Figure 21-9. Structural Model Usage

For the Task II analyses, a more detailed, 3-D model was used to describe the airframe. An isometric view of this model is also included in Figure 21-9.

The wing planform grid for the 3-D model was approximately equal to the 2-D model grid. However, for the 3-D model both upper and lower wing surfaces were represented, including camber and twist of the actual airfoil. Flexible control surface actuators were represented using NASTRAN CELAS elements and MPC equations. The elimination of the X-Y wing midplane of structural symmetry (and load antisymmetry) used on the 2-D models required a redefinition of the location for the wing interfaces with the vertical fin, main landing gear and engines.

The 3-D model fuselage was idealized using 25 frame stations with approximately 10 nodes describing the fuselage half-circumference. NASTRAN bar elements are used to represent fuselage frames with rod elements and quadrilateral shear panels used to represent the fuselage shell.

For both modeling techniques, a network of unit loads, were used on the models to calculate SIC's and to introduce design loads into the model structure. Effective unit load locations are, in general, identical for both models with the exception of the unit loads applied to the 3-D model fuselage which differ from those used on the 2-D models in their application as distributed loads at each frame station. The corresponding structural influence coefficients on the 3-D fuselage model represent an average deflection of the frame node points to which the unit load was distributed.

To assess the results of the structural model techniques and provide insight for future research studies, an investigation was conducted to compare the accuracy (2-D versus 3-D) of the two modeling techniques. This investigation was conducted by up-dating the Task II-A chordwise structural model (2-D), completing a NASTRAN static solution, and comparing these results with those of the Final Design airplane which employed a 3-D structural model. Commensurate with model technique, the input data for the 2-D model reflected the identical coordinate and flexibility data used in the 3-D structural model.

Examples of the results of the 2-D NASTRAN static solution are presented in Figures 21-10 and 21-11 with the corresponding data from the 3-D model included for comparison purposes. These figures show the normalized structural influence coefficients for the wing rear beam and fuselage.

With reference to Figure 21-10, both 2-D and 3-D structural influence coefficients were normalized to the maximum wing tip displacement of the 2-D structural model. This data indicates approximately equal stiffnesses for both models in the basic wing region, inboard of BL 470, with the 2-D model having a more flexible wing tip, e.g., for equal loads applied at the wing tips, the wing tip displacement for the 2-D model would be approximately 10-percent greater than the corresponding displacement of the 3-D model.

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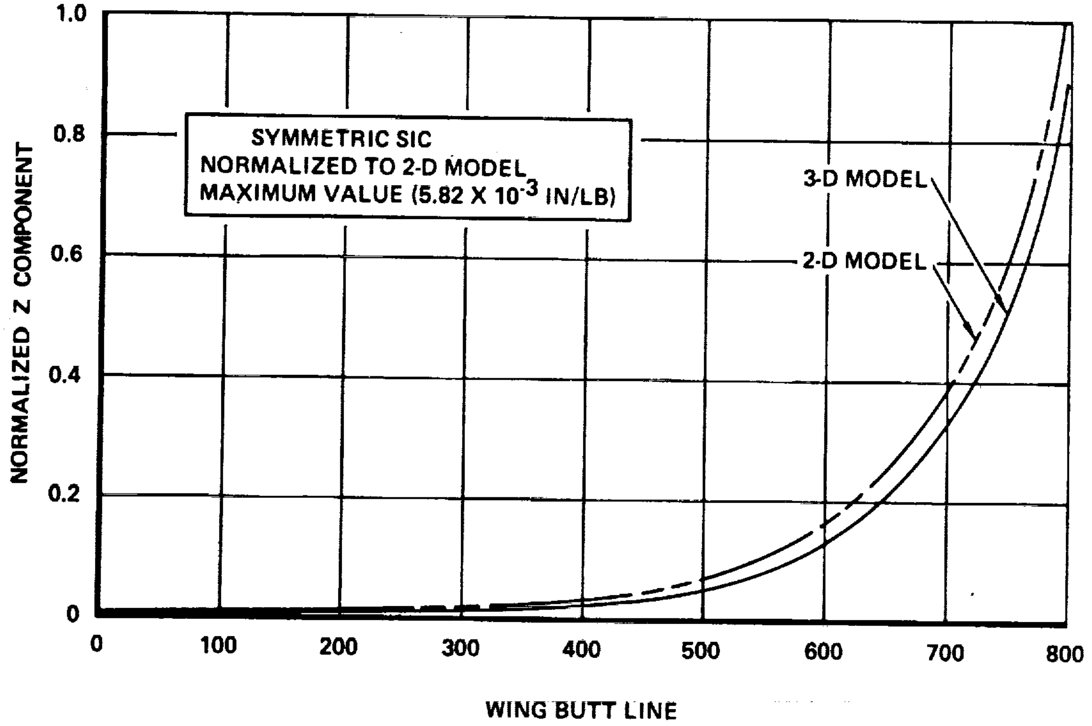

Figure 21-10. Comparison of Wing Rear Beam Structural Influence Lines

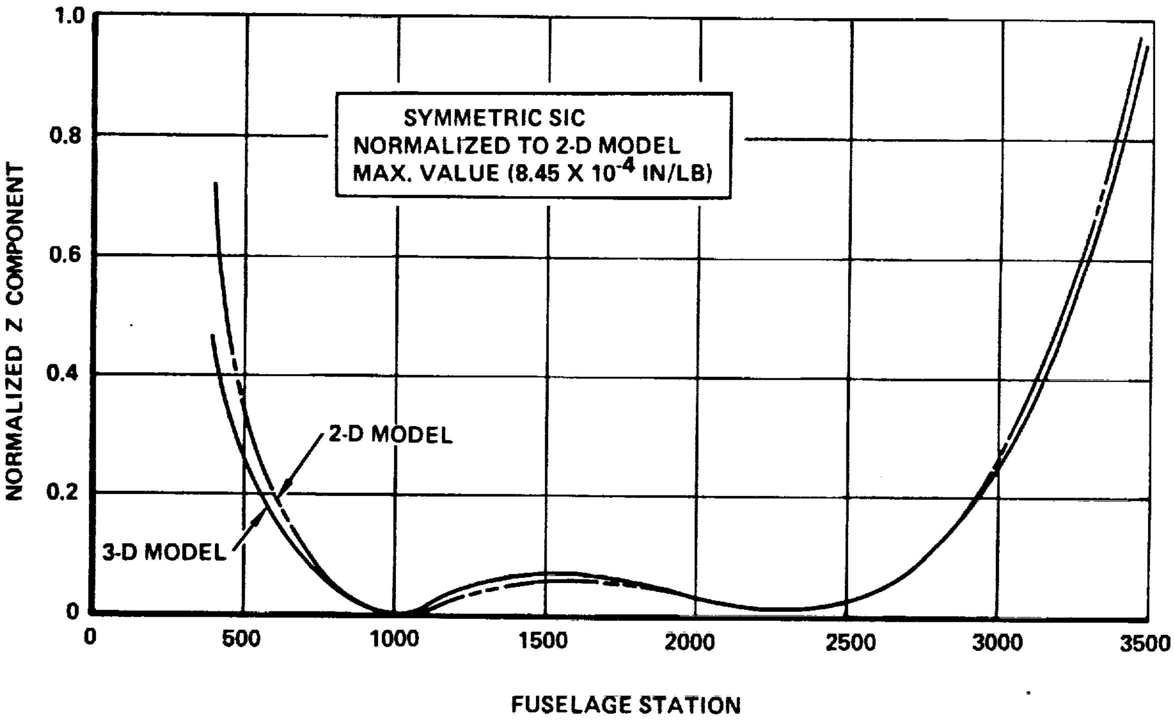

Figure 21-11. Comparison of Fuselage Structural Influence Lines

Similar to the wing results, Figure 21-11 indicates the 2-D model contains a more flexible fuselage forebody region than the 3-D model with a maximum difference in influence coefficients of approximately 50-percent occurring at the nose. Relatively good agreement between the influence lines are noted for the remaining centerbody and aftbody regions.

These results indicate only the relative trends in stiffness associated with each modeling technique; for a complete assessment of the accuracy, additional analyses are required including the overall effects of static aeroelasticity, vibration and flutter, and strength.

Structural Influence Coefficients

A total of 274 generalized coordinates were defined on the structural model for calculating the structural influence coefficients (SIC) matrix and the stiffness matrix. The SIC matrix was used to calculate the aeroelastic loads; whereas, the stiffness matrix after a reduction to approximately 180 degrees-of-freedom was used for the vibration and flutter analyses.

For both the symmetric and antisymmetric boundary conditions, the 274 generalized coordinates were primarily associated with the Z-displacement (vertical) degrees-of-freedom. The exception being the antisymmetric coordinates at the fuselage centerline which were related to the y-displacement (lateral) degrees-of-freedom. In addition, for both boundary conditions, rotational and y-displacement degrees-of-freedom were defined for the wing leading edge and wing vertical, respectively.

Because of the importance of the structural influence coefficients (SIC), an auxiliary plot routine was used to assist in the review of the NASTRAN generated SIC's. In general, this computer program (FAMAS plot routine) extracts the diagonal elements from the NASTRAN SIC matrix for selected sets of wing and fuselage points, normalizes these values to the largest absolute value of each set, and plots these as a function of location (fuselage station, butt line, or water line). These plots display the continuity or discontinuity of the influence lines and were an aid in assessing the validity of the flexibility data input to the structural model.

Internal Loads Runs

NASTRAN internal load runs were conducted on each Task I and Task II model using the aeroelastic loads commensurate with the model stiffness. In addition, the aeroelastic loads for the final design included jig-shape effects which are described in more detail in the following Aeroelastic Loads Analysis section.

The results of the NASTRAN solutions identified the displacements and internal forces (stresses) associated with the elements of each structural model. For the point design stress analysis, the internal loads and stresses from the NASTRAN solution were converted into running loads by an auxiliary FAMAS program.

A comparison of the wing upper surface load intensities (lb/in.) are shown in Table 21-1 for all structural models. These loads were based on the results

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TABLE 21-1. COMPARISON OF WING SURFACE LOAD INTENSITIES - ALL MODELS, MACH 0.90 LOAD CONDITION

PANEL IDENTIFICATION			*LOAD INTENSITY (ULTIMATE), LBS/IN.					
			TASK I			IIA	TASK II B	
REGION	NUMBER	DIRECTION	CHORDWISE	SPANWISE	MONOCOQUE	CHORDWISE	HYBRID (STRENGTH)	HYBRID (FINAL)
WING FORWARD	40322	Nx	-10	-148	-199	-819	-122	-219
		Ny	-1145	-1155	-595	-1120	1109	-1049
		Nxy	201	275	211	143	112	75
WING AFT BOX	40236	Nx	188	122	-925	-377	179	15
		Ny	-10846	-12181	-8102	-11474	-12779	-14311
		Nxy	418	1181	858	436	271	272
	40536	Nx	85	-132	-1483	471	-458	-315
		Ny	-10680	-12318	-8763	-11207	-12680	-14410
		Nxy	1118	2288	2521	1409	1068	1159
	41036	Nx	274	-36	-1094	-567	1052	-1562
		Ny	-6570	-6876	-4544	7040	3522	4725
		Nxy	1369	2027	1949	1581	1583	1773
WING TIP	41316	Nx	701	298	-932	592	-1226	1478
		Ny	-11655	-12546	-8268	-12145	-9504	10106
		Nxy	3492	3240	2528	3773	3686	3730
	41348	Nx	-719	-574	605	-1068	-877	856
		Ny	-6293	5886	-4731	6402	-5148	6598
		Nxy	1535	1797	2132	1990	2290	2608

*LOAD CONDITIONS:

TASK I CONDITION 12: MACH 0.90, n_z = 2.5, W = 700,000 LB, Ve = 325 KEAS
TASK IIA CONDITION 9: MACH 0.90, n_z = 2.5, W = 700,000 LB, Ve 325 KEAS
TASK IIB CONDITION 8: MACH 0.90, n_z = 2.5, W = 700,000 LB, Ve = 325 KEAS

TABLE 21-2. EFFECT OF JIG-SHAPE ON WING UPPER SURFACE LOAD INTENSITIES, TASK IIB HYBRID MODEL

PANEL IDENTIFICATION			* LOAD INTENSITY (ULTIMATE), LBS/IN.		PERCENTAGE DIFFERENCE (%)
			ASSUMED JIG SHAPE		
REGION	NUMBER	DIRECTION	MID-CRUISE	ZERO LOAD	
WING – FORWARD	40322	NX	151	-166	+10
		NY	1106	1083	-2
		NXY	130	124	-5
WING – AFT BOX	40236	NX	67	133	+98
		NY	14650	15596	+7
		NXY	453	499	+10
	40536	NX	1073	1182	+10
		NY	14303	15315	+7
		NXY	1495	1750	+17
	41036	NX	-1812	-2096	+16
		NY	-4220	-4612	+9
		NXY	2106	2471	+17
WING – TIP	41316	NX	1638	1700	+4
		NY	-12407	-14280	+14
		NXY	4009	4262	+5
	41348	NX	1207	1187	-9
		NY	6897	8192	+19
		NXY	2284	2560	+12

* LOAD CONDITIONS:

TASK II B CONDITION 12: MACH 1.25, n_Z = 2.5, W = 690,000 LB, Ve = 294 KEAS

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of the NASTRAN static solution as defined at the six point-design regions used for the structural analysis. The direction of the inplane loads corresponds to the basic airplane axes with the exception of the wing tip regions (panels 41316 and 41318). For these panels the y-direction (spanwise) was parallel to the rear beam of the wingtip structure and the x-direction (chordwise) perpendicular to the rear beam. Conventional sign notation was used; positive signs denote tensile forces and conversely, negative signs denote compression.

A study of the effect of jig-shape on internal loads was conducted using the Task IIB strength-sized model. The structural deflections were determined for one-g flight condition during mid-cruise flight, these deflections were applied negatively to numerically define the jig-shape (i.e., local slope and deformation of the wing grid points). Having established these values the aeroelastic loads were calculated for the 2.5-g symmetric maneuver condition at Mach 1.25. A summary of these results, variation of the point design load intensities with and without the jig-shape effect, is shown in Table 21-2 with the percentage differences indicated.

All load intensities increased when the jig-shape effect was included in the aeroelastic loads calculations; the exception being the spanwise (N_y) and shear (N_{xy}) loads at point design region 40322 which were reduced by 2-percent and 5-percent respectively. As expected, the largest load intensity variations occurred on the more flexible regions of the wing (e.g., a 19-percent increase in spanwise load intensity for region 41348) which is approximately center span of the wing tip.

Because of the large variation in load intensity attributed to the jig-shape effect, the results of this study were incorporated into the element properties for the Final Design F.E. model. In addition, the final aeroelastic loads included the jig-shape effect.

AEROELASTIC LOADS ANALYSIS

Net aeroelastic loads were determined for the aerodynamic shape defined for the baseline configuration at selected conditions on the design flight profile. These loads were calculated using Lockheed's static aeroelastic loads program for inclusion in the NASTRAN internal loads solution, and reflected the specific airframe flexibility of each of the general-types of structural arrangements.

The general logic flow diagram of the aeroelastic loads analysis is presented in Figure 21-12. A more detailed description of the principal tasks involved are as follows:

(1) Determination of the critical flight conditions and their related flight parameters (speed, altitude, acceleration, and gross weight).

(2) Establishment of the basic aerodynamic grid system and local airfoil geometry for calculating the theoretical airloads distributions.

(3) Calculation of the theoretical airloads distribution based on the applicable aerodynamic theory for the flight conditions under investigation. Condense and transfer these theoretical panel point loads

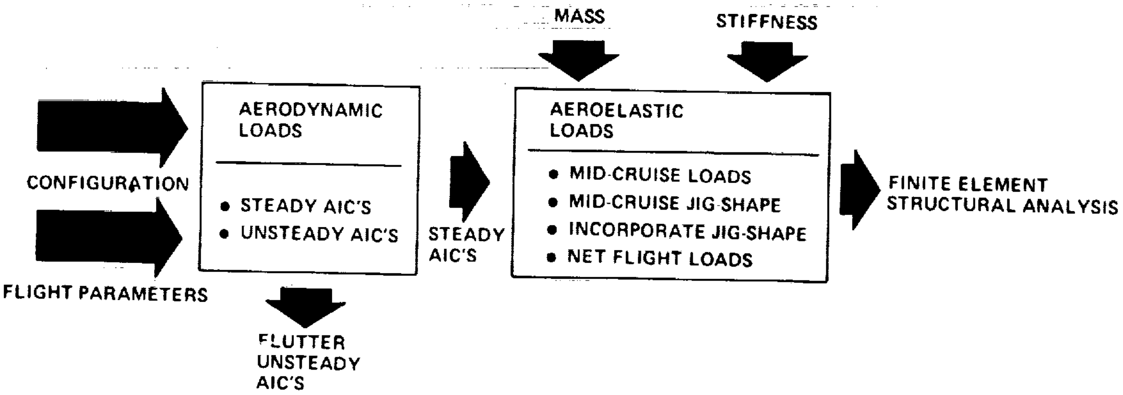
MASS
STIFFNESS
CONFIGURATION
FLIGHT PARAMETERS
AERODYNAMIC LOADS
• STEADY AIC'S
• UNSTEADY AIC'S
STEADY AIC'S
AEROELASTIC LOADS
• MID-CRUISE LOADS
• MID-CRUISE JIG-SHAPE
• INCORPORATE JIG-SHAPE
• NET FLIGHT LOADS
FINITE ELEMENT STRUCTURAL ANALYSIS
FLUTTER UNSTEADY AIC'S

Figure 21-12. Aeroelastic Loads Evaluation Procedure

TABLE 21-3. CRITICAL LOADING CONDITIONS

	CONDITION	MACH NO.	n_Z	SCAT-15F	L-2000-7	969-336C	ARROW WING
Flight	Symmetric	0.30	2.0	-	-	✓	✓
		0.60	2.5	✓	✓	-	✓
		0.90	2.5	✓	✓	-	✓
		1.25	2.5	-	✓	✓	✓
		2.00	2.5	-	-	-	✓
		2.70	2.5	✓	-	✓	✓
		2.90	2.5	-	-	-	✓
	Asymmetric	0.30	0 & 1.67	-	-	-	✓
		0.90		-	-	-	✓
		1.25		-	✓	-	✓
Ground	Taxi	-	2.0	-	-	✓	✓
	Rotation	-	-	-	✓	-	✓
	Landing	-	-	-	-	-	✓

to the structural influence coefficient grid system of the finite-element model and adjust values to reflect any measured steady state lift coefficients and aerodynamic center data when required.

(4) Determination of the net aeroelastic flight loads using the theoretical air load calculated in step (3), the mass matrices, and the airframe stiffness (SIC'S) as defined by the NASTRAN static solution. These data were combined with the aid of matrix algebra to formulate distributed grid point loads on the airplane consistent with the solution of the equations of motions for the prescribed maneuvers.

Critical Flight Conditions

Previous supersonic transport design studies were reviewed to identify potentially critical conditions for the baseline configuration concept. Design conditions for the SCAT-15F, Boeing 969-336C, and Lockheed L-2000-7 supersonic transport are summarized on Table 21-3. Loading conditions evaluated for this study are also included on the table to indicate the scope of potentially critical loading conditions investigated. For the baseline configuration, load conditions were evaluated at both maximum positive and maximum negative load factor and include all conditions where peak values or rapid change in aerodynamic coefficients exist. During the course of this study, all conditions identified on Table 21-3 were investigated sufficiently to assure that critical design loads are included for structural analysis. Supplemental conditions were developed to ascertain the design loads for specific regions of the wing and fuselage.

The loading conditions for the final design cycle included 8 subsonic speed symmetric maneuvers (steady and transient); 7 low supersonic cases, including negative normal acceleration conditions, steady and transient maneuvers at heavy and light gross weights; 4 Mach 2.7 conditions, including mid-cruise level flight and maneuver, and steady and transient maneuvers at start-of-cruise; 2 dynamic gust (pseudo) conditions at Mach 0.90 (positive and negative); and 4 dynamic landing conditions. These load cases are further identified on the design airspeed envelope of Figure 21-15. The gust and landing cases were supplemental conditions developed for the Final Design effort and were selected as critical for fuselage design. The asymmetric accelerated roll condition was not included for the final loads run. The roll case resulted in maximum inplane loads in local regions of the strength-designed wing tip structure. However, with the added stiffness requirements in this region to suppress flutter, the condition was deleted from the list of potentially critical conditions.

Theoretical Aerodynamics

Subsonic and supersonic airloads distribution were determined using the Discrete Load Line Element (DLLE) and Mach Box computer programs, respectively. The DLLE method is theoretically the same as the Doublet Lattice Method of Reference 2; the Mach Box method is described in Reference 3. Typical aerodynamic grids used for determination of subsonic and supersonic aerodynamics are displayed in Figures 21-13 and 21-14.

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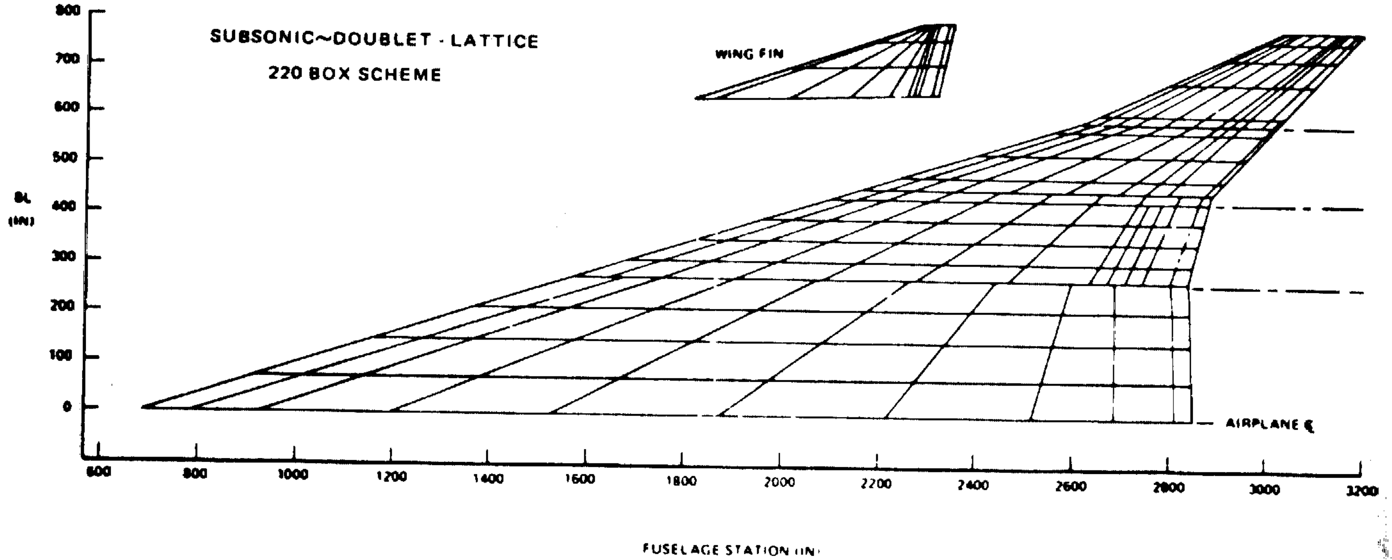
SUBSONIC~DOUBLET - LATTICE
220 BOX SCHEME
WING FIN
BL (IN)
800
700
600
500
400
300
200
100
0
AIRPLANE ℄
600
800
1000
1200
1400
1600
1800
2000
2200
2400
2600
2800
3000
3200
FUSELAGE STATION (IN)

Figure 21-13. Subsonic Aerodynamic Grid

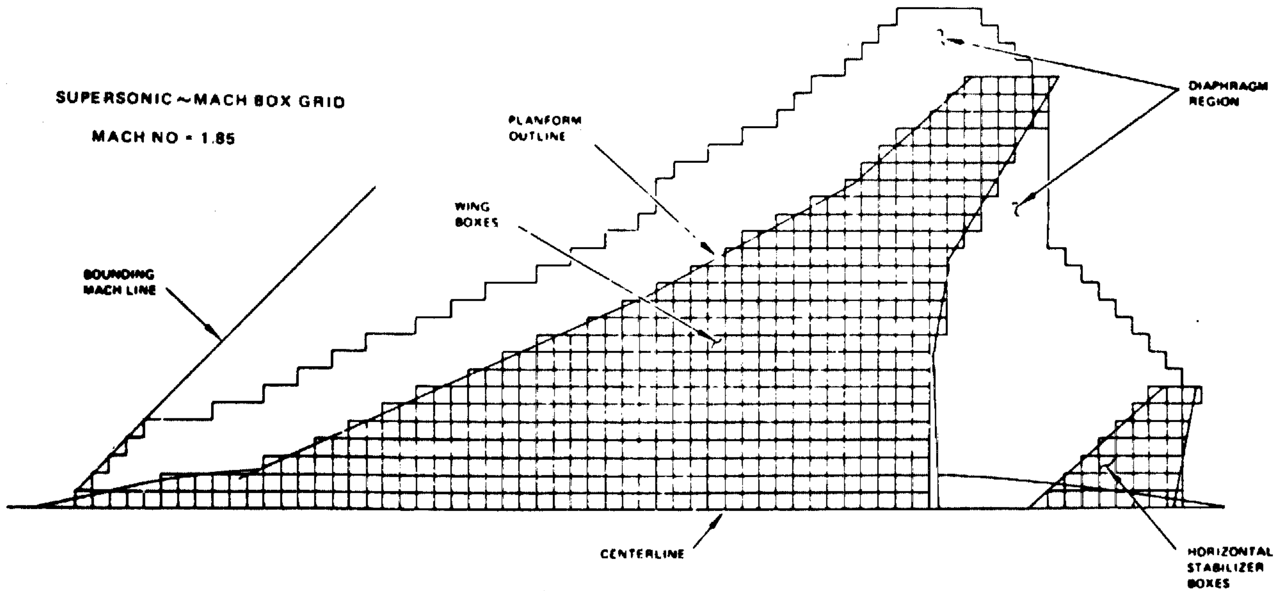
SUPERSONIC~MACH BOX GRID
MACH NO = 1.85
PLANFORM OUTLINE
WING BOXES
BOUNDING MACH LINE
DIAPHRAGM REGION
CENTERLINE
HORIZONTAL STABILIZER BOXES

Figure 21-14. Supersonic Aerodynamic Grid

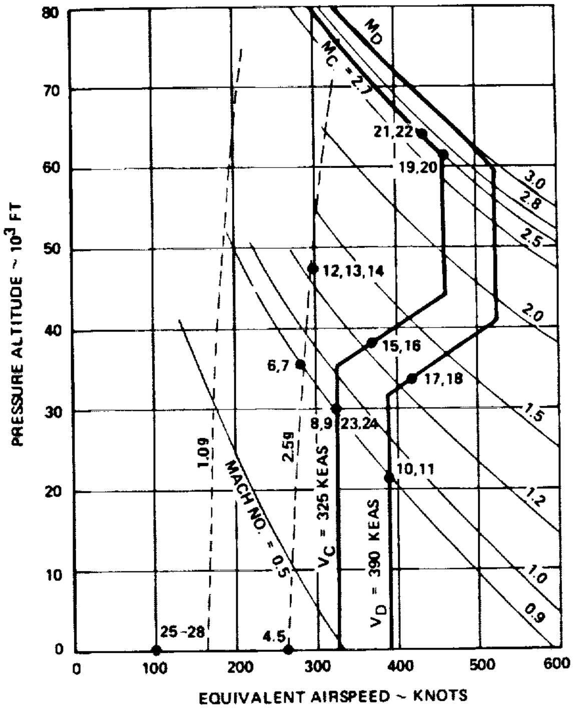

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Figure 21-15. Design Load Conditions - Strength/Stiffness

The above theoretical subsonic and supersonic airload distributions were reviewed for correlation with data developed from the results of NASA wind tunnel tests of the Arrow-Wing configuration.

To assess the net loads effect, static aeroelastic loads were generated for a Mach 2.7 symmetrical maneuver condition using both the wind tunnel measured pressures and the theoretical airload distribution based on Mach box theory. For the airloads based on the wind tunnel data, the pressures on the wing grid system were obtained by interpolation of the corrected data and factored to obtain the lift on each grid element area. These distributions at each angle of attack were combined into a matrix format for application to the net loads program. The data in these matrices were used to define all the airloads on the airplane due to angle of attack. Redistribution of airloads due to flexibility was computed from theoretical aerodynamic influence coefficients.

A comparison of the net integrated wing shears derived by the Mach box method and by the application of the measured pressure data is shown in Figure 21-16. In addition to the wing shears, reduced values of bending moment and torsion at all wing span stations and reduced shears at all fuselage stations were noted for the loads generated using the pressure data. These results occur primarily from the large reduction in tip loading by the measured data causing a significant inboard shift of the spanwise center of pressure.

Reductions in net torsion were less pronounced in the vicinity of the fuselage due to the more forward location of the chordwise center of pressure from measured data at these inboard locations.

Figure 21-17 presents the results of another investigation which was conducted to evaluate the spanwise loading distributions over a wide range of angles of attack at Mach 2.7. Loading distributions were developed using both wind tunnel force and pressure data and a theoretical method using the Mach Box Program.

Span loading distributions from these methods are shown at two airplane angles of attack. The lower angle of attack is within the linear range of wing lift coefficient as a function of angle of attack. For the higher angles, the wind tunnel measured data indicates a significant unloading of outboard wing stations with a high section loading near the fuselage. The Mach Box data represents a wing lift equal to that from integrated pressure data although with a more outboard center-of-pressure location. Available force data indicate a higher wing lift but with the same slope.

Differences between the span loading distributions beyond the linear C_L versus α range are more pronounced with the relationships between the several distributions as previously described. Both force data and integrated pressure data displayed on Figure 21-18 confirm the non-linear trend of wing lift coefficient at higher angles of attack. The Mach Box method is linear and does not display this tendency.

Results of the loading investigations, which were based on limited amount of wind tunnel measured pressure data at Mach 2.7, were inconclusive concerning the choice between using the theoretical aerodynamics or the wind tunnel test data for generating the panel point loads. Even though correlation was not obtained, some geometric similarities were noted between the curves generated using the theoretical aerodynamics and the measured data. For example, approximately equal wing lift slopes were noted for the Mach 2.7 condition using the Mach Box and force data, with higher wing lift values noted for the force data.

Since correlation was not obtained, the design loads for both subsonic and supersonic aerodynamics were based on the applicable theoretical aerodynamics (DLLE and Mach Box) and adjusted to reflect the measured steady state lift coefficients and aerodynamic centers derived from the wind tunnel force data. Table 21-4 presents a summary of the Task I aerodynamic coefficient matrices with their respective wing lift and aerodynamic centers data.

Grid Transform

Grid transforms were required for transferring aerodynamic loadings from the aerodynamic influence coefficient (load point) model to a system compatible with

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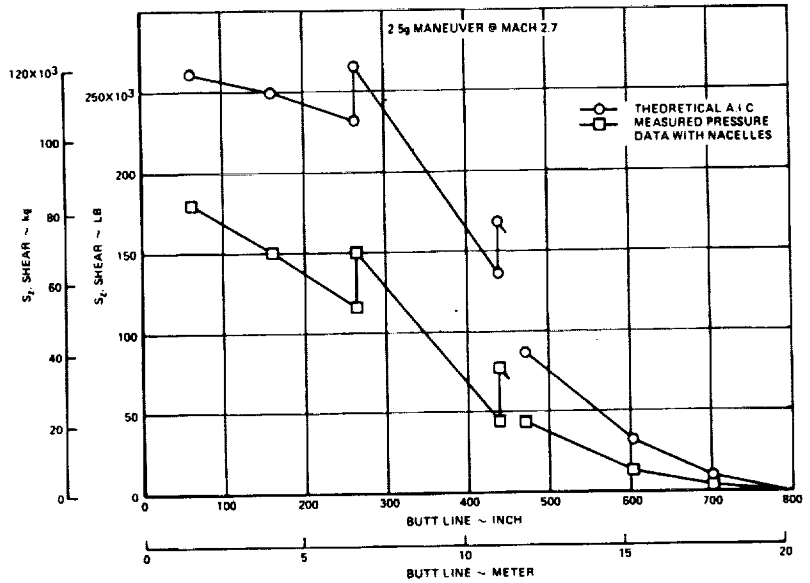


Figure 21-16. Net Integrated Wing Shears - Theoretical vs Measured

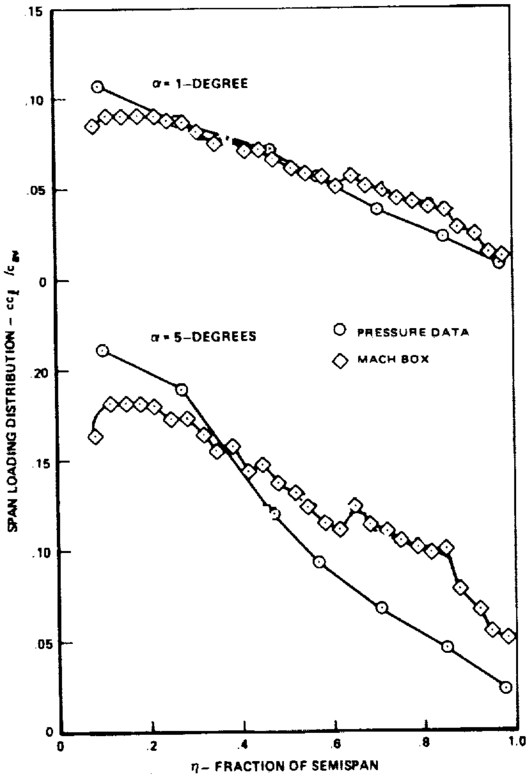


Figure 21-17. Span Loading Distribution - Mach 2.7

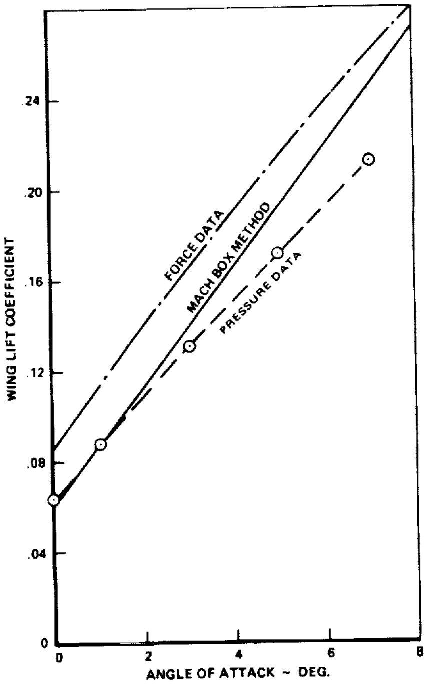


Figure 21-18. Wing Lift Coefficients - Mach 2.7

TABLE 21-4. SUMMARY OF ADJUSTED AERODYNAMIC COEFFICIENT MATRICES — TASK I (k=0)

MACH NUMBER	BOUNDARY CONDITION	MATRIX SIZE	TOTAL EFFECTIVE C_L	LOCATION OF A. C.
0.60	Symmetric	274 x 325	2.42	FS 2324
	Antisymmetric	274 x 233		
0.90	Symmetric	274 x 325	2.58	FS2324
	Antisymmetric	274 x 233		
1.25	Symmetric	274 x 536	2.52	FS 2391
	Antisymmetric	274 x 487		
2.0	Symmetric	274 x 621	1.92	FS 2356
2.70	Symmetric	274 x 621	1.55	FS 2324

the structural influence coefficient (stress and weight) grid. The transformation matrices $[D_z]$ and $[D_\theta]$ express deflection, z, and chordwise slope, θ, respectively, at the AIC points in terms of the deflection, z, at the SIC "given" points. An equivalent set of forces at the structural grid is obtained when the column of lumped aerodynamic forces and pitching moments were premultiplied by the matrices $[D_z]^T$ and $[D_\theta]^T$.

The $[Q]$ matrix, obtained from the aerodynamic influence coefficient program, relates the loading at load points (SIC points) to deflection at control points. Since load points correspond to SIC points on the structural model, a unit $[D_z]$ matrix was used. Determination of angular deflections requires that the deflection at SIC points due to load at control points be defined. One set of control points was used in determining subsonic A.I.C.'s; however, each supersonic speed condition requires a set of control points since their number and location were a function of Mach number.

Mass Distribution

Inertia data were based on the mass distribution commensurate with each of the design stages. These masses were distributed as panel point loads concentrated at the load panel grid, Figure 21-19. This grid system is identical to the Structural Influence Coefficient (SIC) grid system of the structural model. Mass distributions were derived for the operating weight empty condition, and for the payload and fuel, respectively. Using these distributions, which represent one-half airplane values, and the pertinent flight data (n_z, $\ddot{\theta}$, and $\ddot{\phi}$) the inertia loadings at the S.I.C. grid points were determined for the flight conditions investigated. These inertia loading were stored in matrix format for the net loads program.

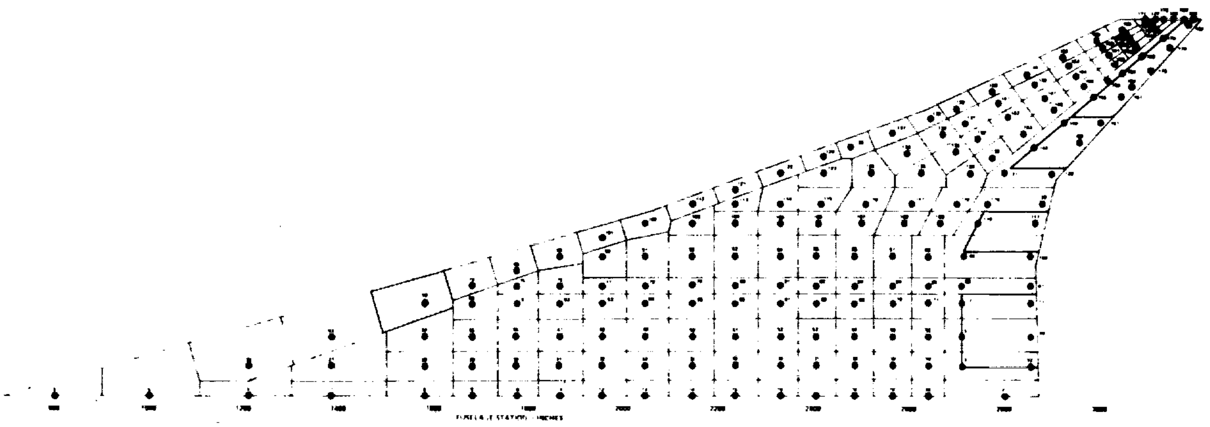


Figure 21-19. Load Panel Grid

Jig Shape Definition

The aerodynamic shape of the aircraft changes during flight due to aerothermoelastic and inertial effects. This is a result of inflight variations in dynamic pressure, Mach number, gross weight, and weight distribution; the latter two result from fuel consumption.

The governing aerodynamic shape serving as the analytical starting point, was the shape providing the optimum performance characteristics in one-g mid-cruise flight. The initial shape of the aircraft was defined at the design cruise lift coefficient by a computer card deck supplied by NASA. This external shape reflected the NASA 15F airplane without a canard or inboard leading edge devices. During the course of this study additional modifications, mainly in the fuselage and wing tip areas, were adopted and incorporated into the configuration as defined by the data deck. Section 2 of this report contains a description of these configuration requirements.

The zero-load shape was designed into the aircraft so that when it was subjected to one-g level-flight loads and to temperatures occurring in the mid-cruise environment, the airframe elastic deformations resulted in an aircraft that had the desired optimum aerodynamic shape. The manufacture of the aircraft was then made in accordance to this zero-load shape in the jig, where the weight was supported in a manner that precludes elastic deformations.

The procedure to establish the jig shape was as follows:

(1) The analytical starting point was the description (camber and twist) of the mid-cruise shape.

$$\left[\alpha_{\text{C \& T, mid-cr}}\right]$$

(2) Analysis was performed to calculate structural deflections due to flight loads occurring during mid-cruise flight. Where the deflection matrix $\left[\Delta\delta_z\right]$ is defined by the product of the structural influence

coefficients [E] and the rigid airplane 1-g loads for the mid-cruise condition.

$$\left[\Delta\delta_z\right] = \left[E\right]\left[P_{z_{\text{1-g mid-cr.rigid}}}\right]$$

Using these calculated deflections and the transform matrix $[D_\theta]$, the incremental changes in chordwise slope and deflection were defined.

$$\left[\Delta\alpha\right] = \left[D_\theta\right]\left[\Delta\delta_z\right]$$

(3) The deflections were applied, negatively, to the mid-cruise shape to establish the jig shape.

$$\left[\alpha_{\text{jig shape}}\right] = \left[\alpha_{\text{C\&T, mid-cr}}\right] - \left[\Delta\alpha\right]$$

The airplane shape used for analytical reference and loft purposes was thereby defined.

Using the Task II strength sized structural model, a test case was computed at Mach 1.25 (V_A) using the jig-shape in lieu of the mid-cruise shape to quantitatively assess the effect of jig-shape on the net external loads and corresponding internal loads. The results of the net loads study indicated a 3-percent increase in bending moment (M_X) at BL. 0 and approximately 10-percent increase at BL. 470. For the internal loads assessment, a matrix of panel point loads was formed and applied to the structural model for conducting an internal loads run. As expected, these results, which are documented in the Finite-Element Structural Analysis section of this section, reflect the same trends as indicated by the integrated loads used to form the net external loads. Based on these results, the aeroelastic loads calculations for the final design (Task IIB) incorporated the above defined jig shape, rather than the mid-cruise shape, for the analytical starting point.

Net Aeroelastic Loads

Net loads for the baseline configuration were formed using Lockheed Static Aeroelastic Loads Program - PSRL F-72. This program permits the aerodynamic influence coefficients to contain moment points and load points in direction other than vertical. Inertia loads were combined with aerodynamic loads to form aeroelastically balanced net loads using the stiffness matrices for each structural arrangement.

Panel point loads were formed into a stacked matrix containing the symmetric and anti-symmetric load components for each flight condition.

VIBRATION AND FLUTTER ANALYSIS

The logic flow schematic depicting the principles of the analytical procedure applied to flutter analysis and optimization of the baseline configurations is presented in Figure 21-20 and is discussed in more detail in the following text.

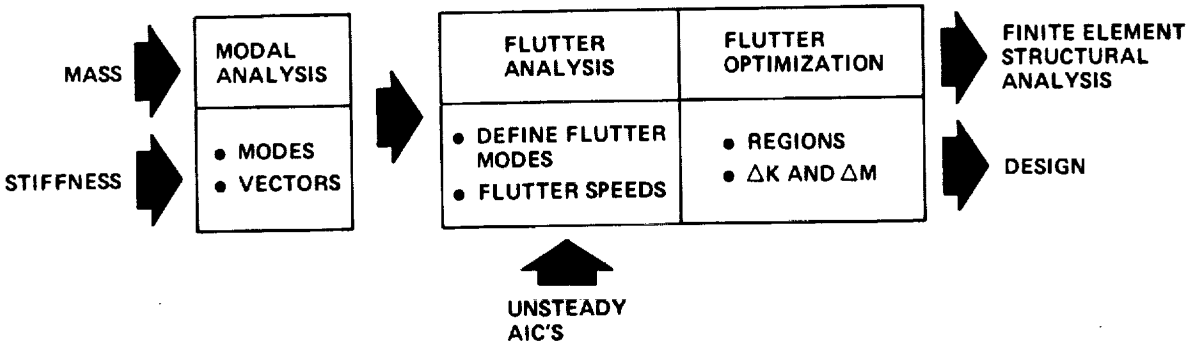


Figure 21-20. Vibration and Flutter Analysis

Structural Model

The vibration and flutter investigation used the Finite-Element Structural Model as the basis for formulating the analytical math model for each structural arrangement. Thus, these math models reflect the modeling technique, airframe stiffness, and mass associated with each of the structural models. A detail discussion of the structural models is contained in the previous section entitled "Finite-Element Structural Analysis".

The math models used in the vibration analyses employed a coordinate system and associated degrees-of-freedoms that related directly to the structural model structural influence coefficients grid. These degrees of freedom are of the 188th order symmetrically, Figure 21-21, and 178th order antisymmetrically. For both boundary conditions, the degrees of freedom are mainly associated with the Z-Axis displacements but includes Y-Axis displacements for the wing vertical fin. For the antisymmetric boundary, the fuselage degrees of freedom are related to Y-Axis displacements.

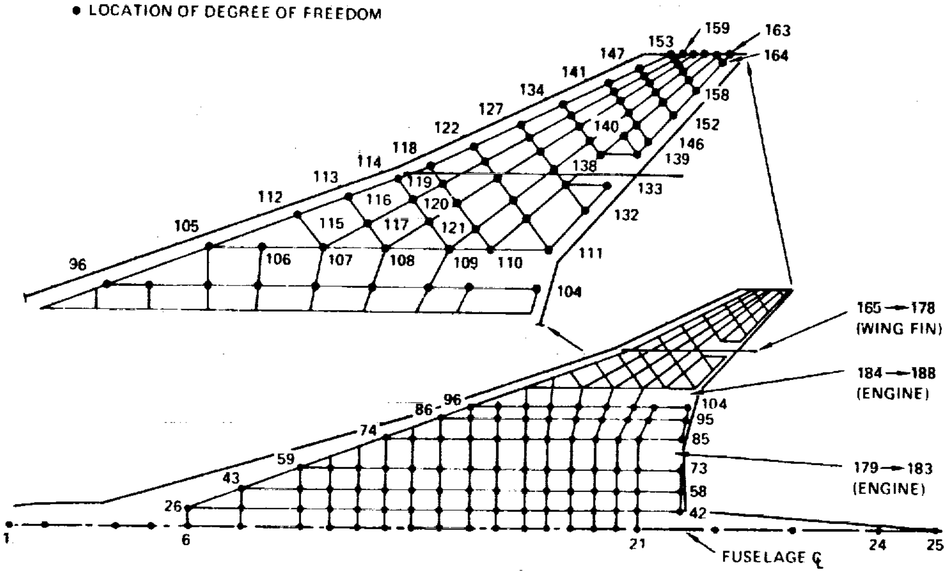
The stiffness matrices were obtained by condensing (Guyan Reduction) the large-order stiffness matrices generated by the NASTRAN static solutions to a size that is conformal with the symmetrical and antisymmetrical degrees of freedom.

The choice of the number of symmetric degrees of freedom was based on the constraint that a 188th order vibration problem was the maximum size that could be run in the Lockheed FAMAS computer system. It is felt though that a 188th order problem retains adequate structural definition and still gives good visibility for model trouble shooting and verification.

Vibration Analysis

Vibration modes were calculated using several different approaches and results compared to select the analytical method for application to the design concepts study, Table 21-5. The three methods investigated were: Inverse Power (INV) and Givens (GIV) method available in NASTRAN and the FAMAS QR method.

The first eigenvalue routine executed was the Inverse Power (INV). The approach was to take advantage of the sparseness of the stiffness and inertia matrices in the F-set (858th order) and solve for a limited number of modes. Computer time for the INV method was 80.5 seconds of CPU time per mode.



● LOCATION OF DEGREE OF FREEDOM
159
163
164
153
147
141
134
127
122
118
114
113
112
105
96
158
152
146
139
140
138
133
132
119
116
120
115
117
121
106
107
108
109
110
111
104
165 → 178 (WING FIN)
184 → 188 (ENGINE)
179 → 183 (ENGINE)
96
86
74
59
43
26
1
6
21
104
95
85
73
58
42
24
25
FUSELAGE ℄

Figure 21-21. Symmetric Degrees of Freedom for Vibration Analysis

Because of the very high computer times, the problem was reduced to the 188th order. This matrix reduction required 89 seconds. The INV method was again executed and resulted in 60.5 seconds of CPU time per mode; whereas, the Givens method for the same reduced order problem resulted in 2.3 seconds of CPU time per mode.

The reduction to 188th order eliminated only a few inertial degrees of freedom (approximately 10-percent) and resulted in a problem size small enough so that the FAMAS QR method could also be exercised. The QR method resulted in 3.6 seconds of CPU time per mode.

The Givens and QR methods not only result in a marked reduction in computer time but also solve for all 188 roots, a definite advantage over the INV.

TABLE 21-5. ANALYTICAL METHODS FOR VIBRATION ANALYSIS

METHOD	MATRIX SIZE	NUMBER ROOTS	CPU TIME (SEC)		FREQUENCY (2) (HZ)	
			TOTAL	PER MODE (1)	MODE 1	MODE 2
INVERSE POWER	858	20	1610	80.5	1.470223	2.049600
INVERSE POWER	188	2	121	60.5	1.471204	2.055224
GIVENS	188	188 (40 VECTORS)	83	2.1	1.471203	2.055210
GIVENS	188	188 (40 VECTORS)	93	2.3	0.9275528	1.007738
QR	188	188 (40 VECTORS)	145	3.6	0.9275518	1.007738

(1) BASED ON NUMBER OF EIGEN VECTORS FOUND
(2) FREQUENCIES 1 AND 2 ARE NOT NECESSARILY THE LOWEST FREQUENCIES. ALSO THE LAST TWO CASES (GIVEN AND QR) USED A DIFFERENT MASS MATRIX THAN THE FIRST THREE CASES

method where it is possible to miss a mode. No significant differences in accuracy was noted between the three methods.

Based on the results presented the Givens Method is superior to the other methods and was selected as the vibration analysis method for the study.

The stiffness matrices of each structural arrangement, as derived from the structural models, were combined with the appropriate inertia matrices to compute the symmetric and anti-symmetric eigenvectors and eigenvalues of the free-free airplane. The inertia matrices were formed for two airplane weight conditions namely: the operating weight empty (OWE), and the full fuel and full payload (FFFP). These weight conditions represent the extremes of minimum and maximum weight. No intermediate weight conditions were examined. In general, 50 vibration modes were extracted from each vibration solution for use in the flutter analysis and flutter optimization.

Aerodynamic Formulation

The steady and unsteady aerodynamic influence coefficients (AIC) were computed for Mach 0.60, 0.90, 1.25 and 1.85. The flutter analyses conducted on the Task II final design airplane included an analysis of these Mach numbers with the exception of the Mach 1.25 condition; whereas, the Task I analytical design studies encompassed only the first three Mach numbers. The Mach 0.60 and 0.90 AIC's were computed by the Doublet-Lattice method of Reference 2, while the Mach 1.25 and 1.85 AIC's were computed by the Mach Box method of Reference 3. The AIC's were computed for the wing, the wing fin, and the empennage surfaces, and were adjusted, when required, to reflect measured wind tunnel force data steady state lift coefficients and aerodynamic centers. The Mach 0.60 and 0.90 AIC calculations account for the interference between the wing and the wing fin; whereas, the Mach 1.25 and 1.85 AIC's do not include this effect.

The significance of the aerodynamic interference between the wing and wing fin is shown by Figure 21-22, which presents $C c_\ell / C C_{L\alpha}$ and aerodynamic center (a.c.) versus fraction of the semispan. The data visually relates the wing fin interference effect on the distribution for the applicable Mach numbers. For the Mach 0.60 and 0.90 conditions, which include the interference effect, an increase in the $C c_\ell / C C_{L\alpha}$ distribution inboard of the wing fin and a decrease in the $C c_\ell / C C_{L\alpha}$ distribution outboard of the wing fin is noted in contrast to the clean-wing aerodynamics calculated for Mach 1.85.

The normalized $C_{L\alpha}$ versus reduced frequency is presented in Figure 21-23. The normalized $C_{L\alpha}$ is $C_{L\alpha}$ at a finite reduced frequency divided by $C_{L\alpha}$ for a reduced frequency of zero. The figure presents the real and imaginary parts of the normalized $C_{L\alpha}$ for Mach 0.90 and 1.85 and thus shows the variation in amplitude/phase as a function of Mach number and reduced frequency. As can be seen the Mach 1.85 aerodynamics is composed primarily of the real part. This differs from the Mach 0.90 aerodynamics which exhibits a mix of the real and the imaginary parts.

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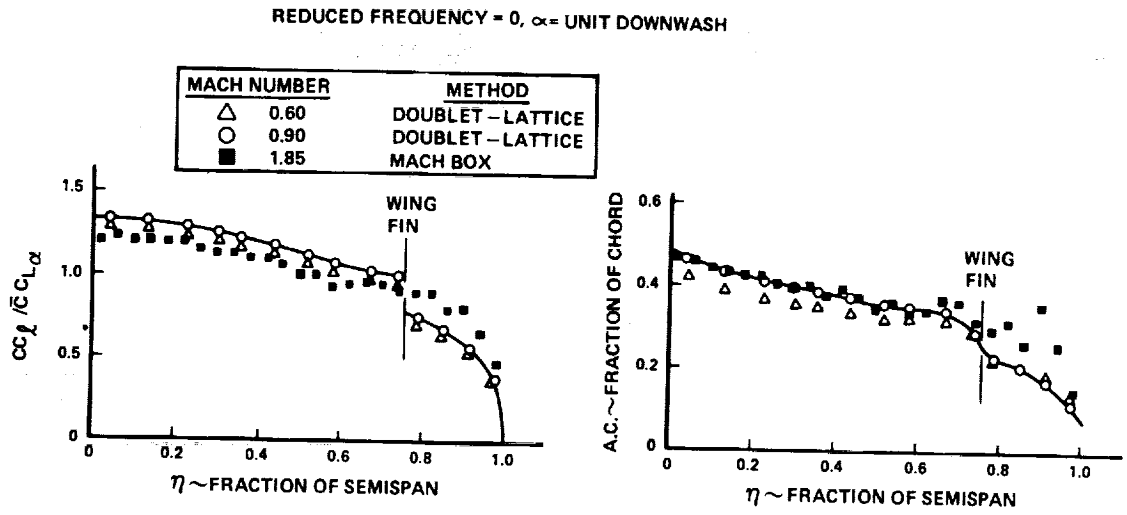

NOTE: THE WING/WING FIN AERODYNAMIC INTERFERENCE EFFECT:
- INCLUDED IN THE MACH 0.60 AND 0.90 AERODYNAMICS
- NOT INCLUDED IN THE MACH 1.85 AERODYNAMICS

Figure 21-22. Theoretical Aerodynamic Distribution

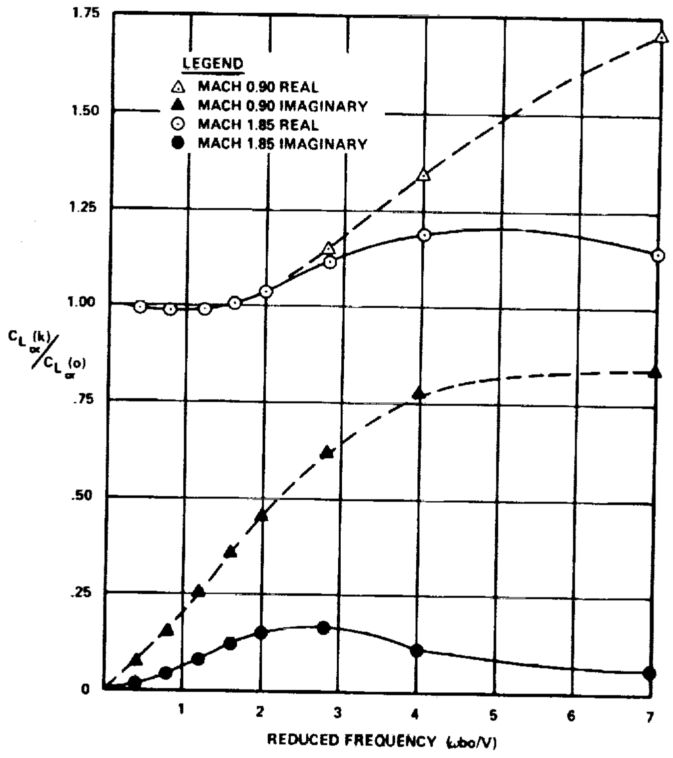

Figure 21-23. Normalized C_L Versus Reduced Frequency

Flutter Analysis

The flutter analyses of each of the structural arrangements investigated for application to the arrow-wing configuration was conducted using the method of solution described in Reference 4 as the p-k method. This method is contained in the FAMAS library and results in a solution which defines rate of decay and frequency for preselected values of speed and gives matched altitude, Mach number, and reduced frequency (k) for each mode at each preselected velocity.

All matrices involved in this solution are real and uniquely defined, except for the aerodynamic influence coefficient matrix which is complex and must be given for a sufficient number of k values. For the p-k method, the flutter equation is solved at several values of airspeed and air density, or combinations thereof, for complex roots p associated with the modes of interest. These modes of interest were determined from a review of the vibration analysis and/or from previous flutter analysis. In addition, all analyses assume a structural damping of 2-percent.

Two fundamental questions arose during the flutter analysis, namely:

(1) How many vibration modes are required to arrive at the converged flutter solution?

(2) How many AIC matrices, as a function of reduced frequency (k), are required to arrive at a converged flutter solution?

Both of these questions were investigated during the course of the flutter studies.

In response to the first question, velocity versus damping for the 10, 15 and 20 vibration mode flutter analyses are shown in Figure 21-24. It can be seen that the character of the flutter modes can be significantly changed by going from 20 to 10 vibration modes. Figure 21-25 shows the flutter velocity of the bending and torsion flutter mode as a function of the number of vibration modes used in the flutter analysis. This figure shows that the flutter velocity changes only 1-percent when the number of vibration modes vary from 20 to 50. As a consequence of this study, 20 or more vibration modes were used in all subsequent flutter analyses.

The number of AIC matrices required to arrive at a converged flutter solution was investigated by running a flutter analysis with AIC matrices corresponding to 17 k values and then repeating this analysis with every other AIC matrix eliminated (9 k values). Within the reading accuracy of the flutter plots the results from these analyses were identical. As a consequence of this study, AIC matrices corresponding to at least 9 k values were used in all subsequent flutter analyses.

Symmetric and antisymmetric flutter solutions were conducted at sufficient Mach numbers to assess the structural dynamic characteristics of the various structural arrangements. And in general, these solutions were conducted for both operating weight empty (OWE) and full fuel and full payload (FFFP) airplane weight conditions. As an example of these solutions, the symmetric flutter solution for the OWE Final Design airplane at Mach 0.90 is shown in Figure 21-26. For this condition, the critical flutter mode was the wing bending and torsion mode. The figure also presents a trace of the critical flutter speed for this mode on

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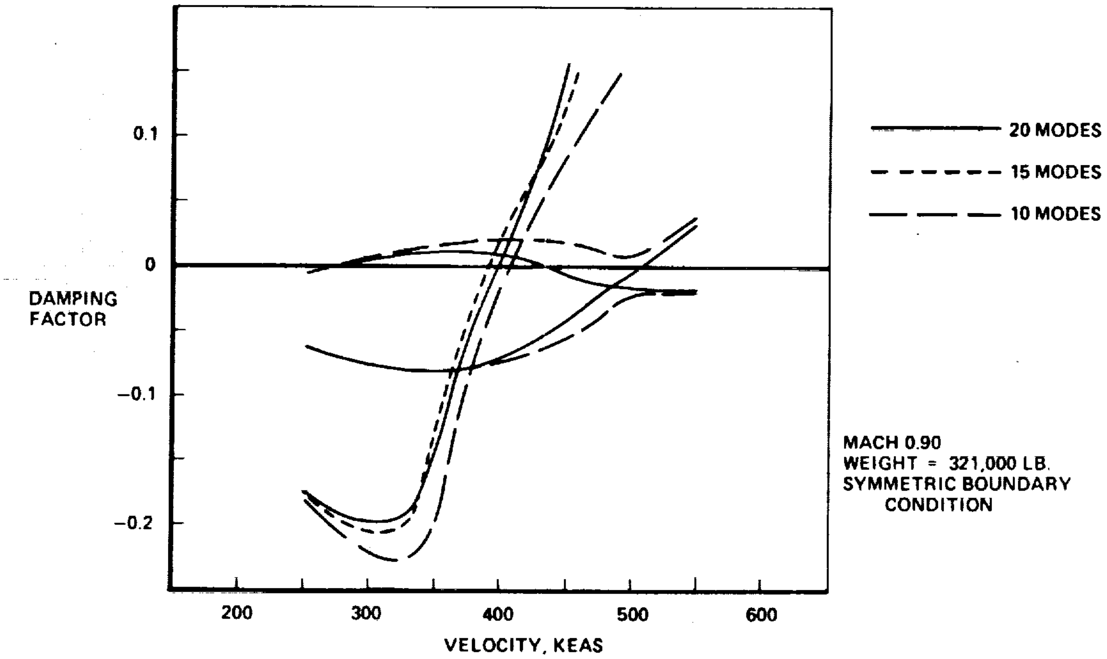


Figure 21-24. The Effects of Number of Modes on the Flutter Solution

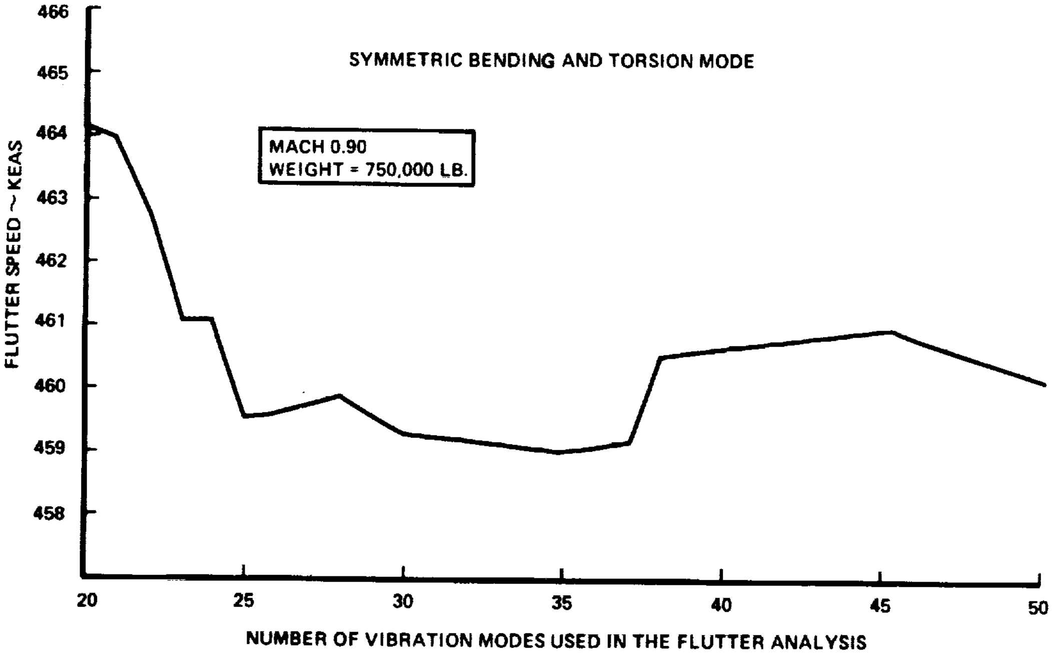
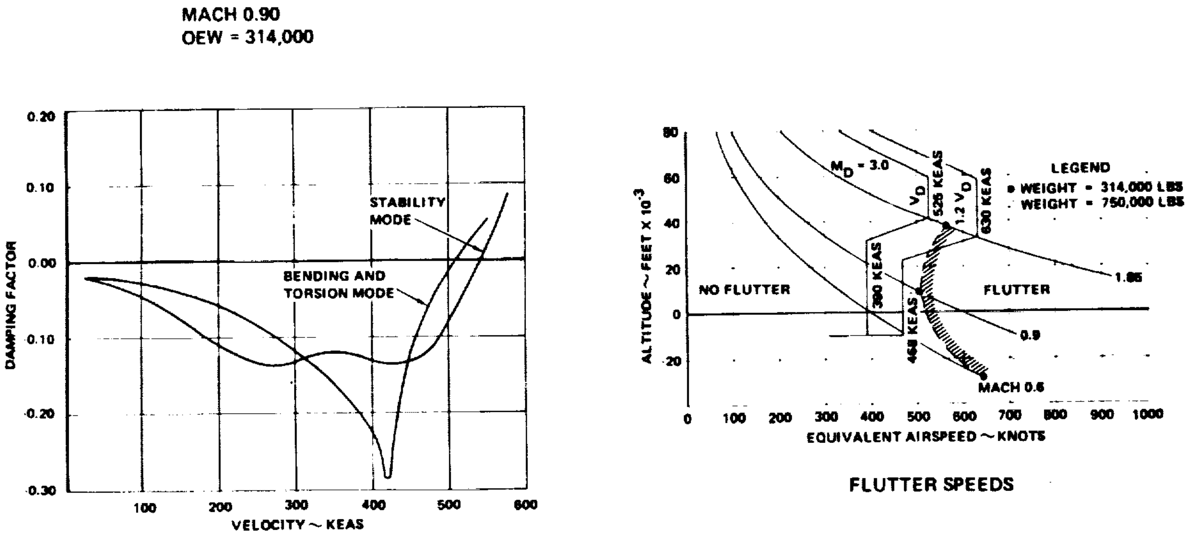


Figure 21-25. Flutter Speed Variation With Number of Modes



FLUTTER SPEEDS

FLUTTER ANALYSIS

Figure 21-26. Flutter Analysis Results

SYMMETRIC FLUTTER ANALYSIS - CHORDWISE STIFFENED ARRANGEMENT

MACH NO. = 0.90 WEIGHT = 750,000 LBS.

PARTICIPATION COEFFICIENTS - FLUTTER MODE 8 - BENDING AND TORSION MODE

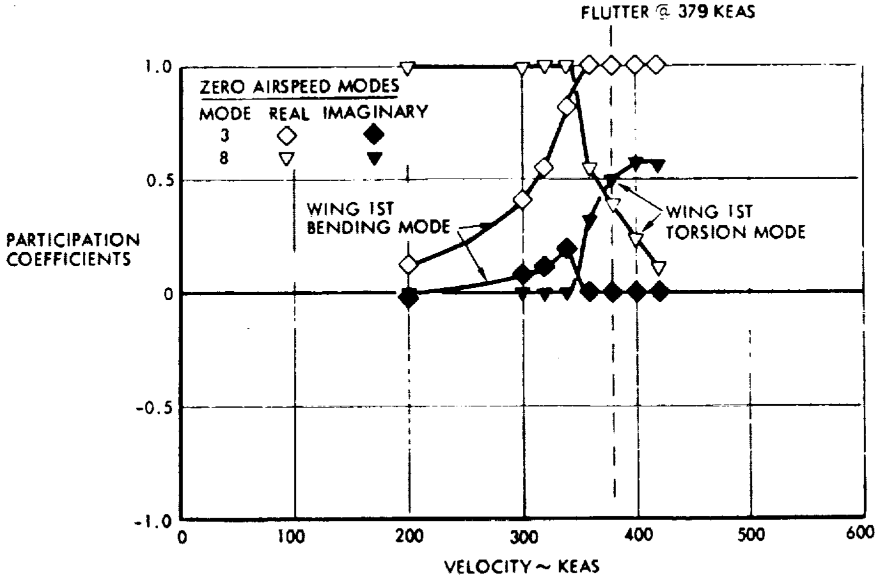


Figure 21-27. Participation Coefficients - Mach 0.9 - FFFP-Mode 8

the design flutter envelope. Similar critical speed traces were constructed for each flutter mode to completely define the flutter boundary.

In general, three distinct flutter mechanisms were noted throughout this investigation; they were: the bending and torsion mode, the hump mode, and the stability mode. Considerable insight into the modal composition or these flutter mechanisms was provided by reviewing the participation coefficients. Participation coefficients are the complex eigenvectors associated with the roots of a flutter solution.

As an example of this technique, the participation coefficients for the bending and torsion flutter mode (mode 8 of Figure 21-28) reveal that at flutter, this mode is principally composed of the zero airspeed Modes 3 and 8 (Table 21-6). Participation coefficients resulting from the symmetric flutter analysis at Mach 0.90 for the full fuel and full payload (FFFP) is shown in Figure 21-27. As indicated on this figure, the zero airspeed wing 1st bending mode rapidly transitions through the adjacent higher frequency modes and couples at flutter with the zero airspeed wing 1st torsion mode. This conclusion is not obvious by reference to the frequency velocity diagram of Figure 21-28.

To understand the flutter mechanisms of the arrow-wing configuration more thoroughly, a flutter analysis was conducted with the wing rigid inboard of BL 470, i.e., flexible wing tip. This investigation was conducted for the Mach 0.90 condition using the Task I chordwise-stiffened structural model for the 750,000-lb aircraft, (FFFP). The symmetric flutter analysis of this configuration showed that for Mach 0.90, the wing 1st bending mode rapidly increases in frequency with increasing velocity and coalesces with the wing 1st torsion mode to flutter at 418 keas. This flutter mechanism was identical to the flutter mechanism for the flexible aircraft. For the unrigidized or flexible aircraft the bending and torsion mode flutter velocity was 379 keas.

Flutter Optimization

An interactive computer graphics program was utilized in the optimization of the arrow-wing configuration. An abbreviated description of the equations, method of solution, and optimization procedure are presented in Reference 5. The general steps of the overall procedure are listed below, followed by a more detailed description of the flutter optimization process.

- Define the critical flight condition and associated flutter modes from a review of the basic vibration and flutter analyses.
- Establish the basic design regions and corresponding design variables for application to the flutter optimization process.
- Calculate a sufficient number of incremental stiffness matrices $[\Delta K]$ and associated mass matrices $[\Delta M]$ to cover the expected range of investigation.
- Operate the flutter optimization program to define the added mass (weight penalty) associated with attaining the design flutter speed.

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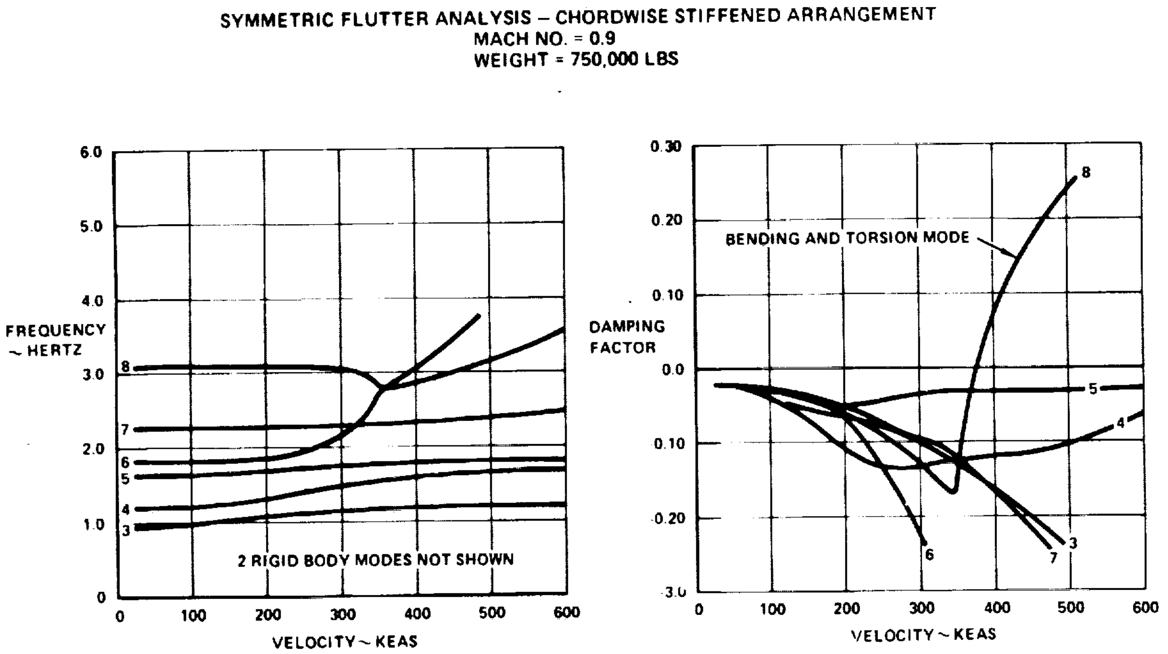

Figure 21-28. Symmetric Flutter Analysis - Mach 0.9 - FFFP

TABLE 21-6. LOWER FREQUENCY SYMMETRIC VIBRATION MODES - CHORDWISE STIFFENED

MODE NUMBER	MODE DESCRIPTION	MODE FREQUENCY ~ HERTZ	
		OWE	FFFP
1	RIGID BODY	0.000	0.000
2	RIGID BODY	0.001	0.001
3	WING 1ST BENDING	1.009	0.933
4	FUSELAGE 1ST BENDING	1.381	1.206
5	ENGINE PITCH IN PHASE	1.641	1.627
6	ENGINE PITCH OUT OF PHASE	1.817	1.815
7	FUSELAGE 2ND BENDING	2.784	2.261
8	WING 1ST TORSION	3.288	3.104

OWE ~ WEIGHT = 321,000 LBS
FFFP ~ WEIGHT = 750,000 LBS

Basic Design Regions.- For the initial optimization studies conducted in Task I, the selection of design regions was of an exploratory nature to provide a general assessment of the effectiveness and optimum distribution of material for the overall wing planform. Thus, the wing planform was divided into 10 general regions (Figure 21-29), which included the two engine support beam locations. Natural boundaries were retained in the establishment of these regions as indicated by their location with respect to the landing gear well, major chordwise ribs, wing vertical, etc.

The application of the coarse-grid modeling philosophy to the optimization procedure provided the necessary insight for the investigator to eliminate the regions which least affected the critical flutter mechanism under investigation. Thus, a more detailed grid system, with fewer overall regions, can be concentrated at the most efficient wing location. As an example, the solution for the Task I monocoque design, which used the coarse grid system shown in Figure 21-29, indicated the wing tip structure to be the most effective region for adding stiffness and mass to achieve the desired flutter speed. It was also shown that the bending and torsion mode flutter mechanism was controlled by the wing inertial and flexibility characteristics outboard of BL 470. Thus, for the more detailed evaluation of the stiffness requirements for the Final Design airplane the optimization effort focused on the wing tip structure. Five design regions were defined for the wing tip structure planform (in lieu of 2 for Task I) as indicated on Figure 21-30. The establishment of the design region boundaries considered the location of the wing vertical and appropriate ribs and spars as may be required. However, the primary influence in the selection was the natural boundaries defined by the structural model.

Design Variables.- The flutter optimization procedure uses incremental stiffness and mass parameters as a set of structural design variables. Using these parameters, the expressions for mass and stiffness take the following form:

$$\left[M\right] = \left[M_0\right] + \sum_{i=1}^{n} \beta_i \left[M_i\right]$$

$$\left[K\right] = \left[K_0\right] + \sum_{i=1}^{n} \beta_i \left[K_i\right]$$

where $[M]$, $[K]$ are the matrices of total mass and stiffness; $[M_0]$, $[K_0]$ are matrices of the fixed mass and stiffness; $[M_i]$ and $[K_i]$ are mass and stiffness matrices associated with a unit of the design variable β. The design variable β can be a unit of area or thickness as related to their respective axial element or shear panel element defined in the structural model.

For the flutter calculations, a reduction in the size of stiffness matrix is required over the large-order stiffness matrix obtained from the structural model. As a result of this coordinate reduction, the relationship between the design variable β and its associated stiffness K are nonlinear and require additional computer runs to define this relationship with sufficient accuracy. Data points to define this nonlinear relationship are derived by formulating a

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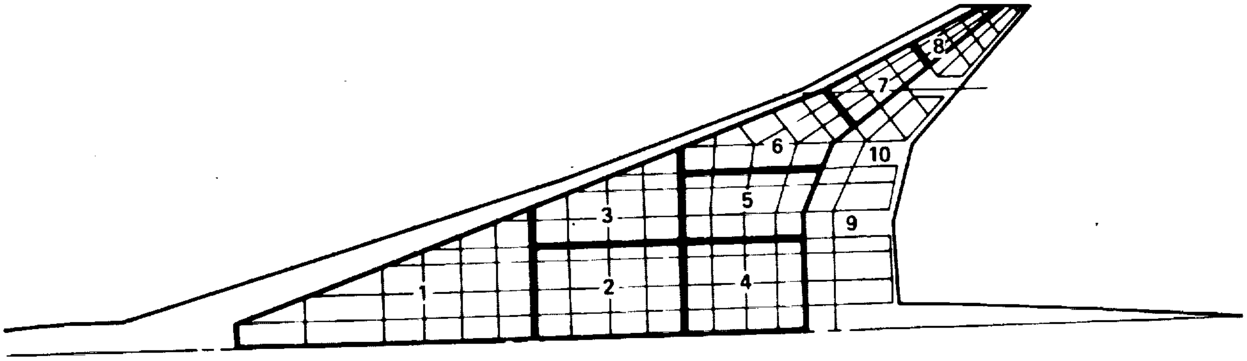
1
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10

Figure 21-29. Design Regions for Flutter Optimization - Task I

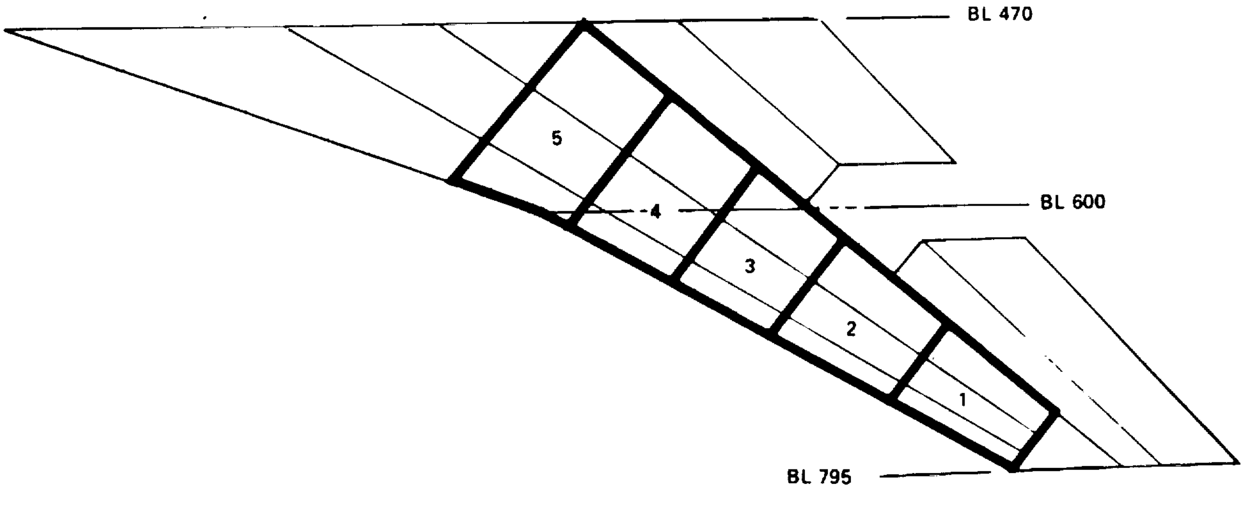
BL 470
BL 600
BL 795
5
4
3
2
1

Figure 21-30. Design Regions for Flutter Optimization - Task II

separate NASTRAN bulk data deck for each of the basic design regions. These decks contain the element property cards with the incremental changes in geometry. Using these decks, the stiffness matrices are obtained from the NASTRAN static solution and condensed to the desired size for the flutter optimization process.

An example of this effect is shown in Figure 21-31, which shows the increments in four elements of the Arrow Wing stiffness matrix as a function of increments in cover sheet thickness of the appropriate design variable. The stiffness matrices from which these results were obtained are from the Task I chordwise-stiffened arrangement.

Optimization Procedure.- Once the basic design regions were established the optimization process was initiated. This involved the following steps:

(1) The basline stiffness, $[K_0]$, and mass matrices, $[M_0]$, were formed using NASTRAN in a large order system. $[M_0]$ was input as a matrix in the NASTRAN Bulk Data Deck.

(2) These matrices were condensed to 188th order using the Guyan Reduction so that a vibration case could be more readily obtained.

(3) The vibration calculation which was then performed produced 50 normal mode shapes. These can be used to reduce the problem size to 50th order by a pre- and post-multiplication of the baseline 188th order $[K_0]$ and $[M_0]$.

(4) A flutter solution was then obtained using these matrices. This yielded the flutter speed for the baseline configuration.

(5) NASTRAN was used to generate $[\Delta K]$ and $[\Delta M]$ matrices. Only the increment in element sizing for a particular $[\Delta K]$ was input for the structural model. The base stiffness, $[K_0]$, was then added to $[\Delta K]$ in the large order system, and the resulting $[K_0 + \Delta K]$ was condensed by the Guyan Reduction to 188th order. The NASTRAN mass generator program was used to form $[\Delta M]$ directly from the sizing increments (Δt) on the element property cards in the NASTRAN Bulk Data Deck. This matrix was then reduced to 188th order by the same technique. $[\Delta K]$ matrices in this system were formed by subtracting $[K_0]$ from $[K_0 + \Delta K]$.

(6) These $[\Delta K]$ and $[\Delta M]$ matrices were reduced to 50th order by modalizing with the normal mode shapes from the basline vibration analysis.

(7) All of these 50th order matrices were input into an interactive computer graphics flutter optimization program. This program allows the engineer to choose a flutter speed and then calculate β- "how much" of each $[\Delta K]$, taken one at a time, is necessary to satisfy this constraint. The engineer may update the structure to any combination of $[\Delta K]$ and $[\Delta M]$ matrices desired. In doing this, the β solutions are taken into account, and the structure is modified so that an optimum structure (minimum weight) results.

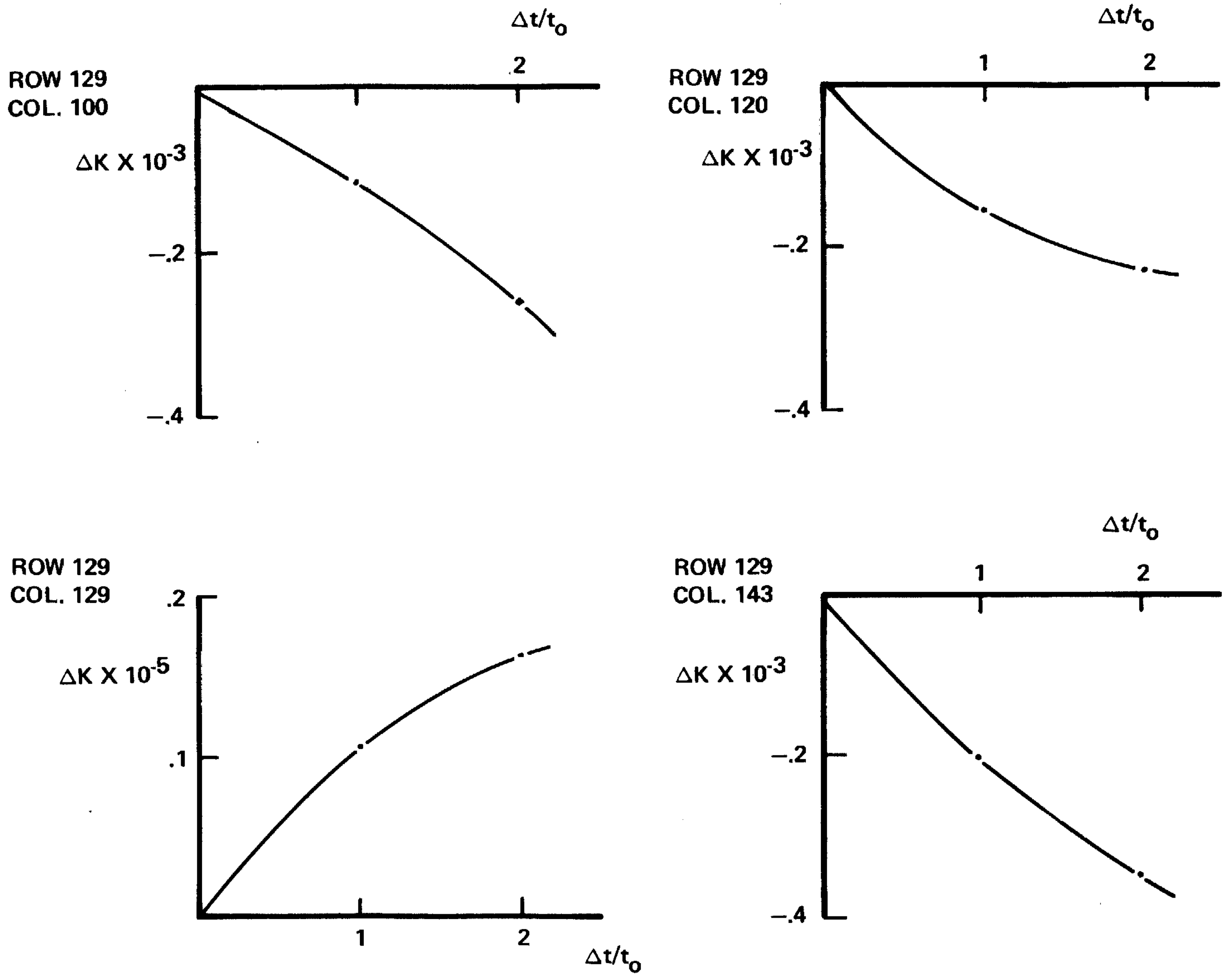

Figure 21-31. Nonlinearities in the Incremented Stiffness Matrix Elements, Arrow Wing

(8) Because the modifications made to the structure above were not reflected in the vibration modes used in the analysis, it was necessary to define a new baseline configuration. Thus the resulting structure from (7) was defined as a new baseline configuration and the process was repeated in an iterative manner until the distribution of β's converged on a solution and gave the desired flutter speed. This was the optimum design for flutter.

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